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PREPRINT TASK I

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ANALYSIS OF SELECTED DEEP-SPACE MISSIONS

A REPORT COVERING TASK I EFFORT
UNDER THE STUDY OF

NASA EVALUATION WITH MODELS OF OPTIMIZED NUCLEAR SPACECRAFT (NEW MOONS)

Wm. S. West, J. Michael L. Holman, and Herbert W. Bilsky

APRIL 1969

GSFC GODDARD SPACE FLIGHT CENTER
GREENBELT, MARYLAND

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TASK I

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NASA Evaluation With Models Of Optimized Nuclear Spacecraft
(NEW MOONS)

Wm. S. West, Technical Officer
Goddard Space Flight Center
Greenbelt, Maryland

and

J. Michael L. Holman and Herbert W. Bilsky
RCA
Astro-Electronics Division
Defense Electronic Products
Princeton, New Jersey

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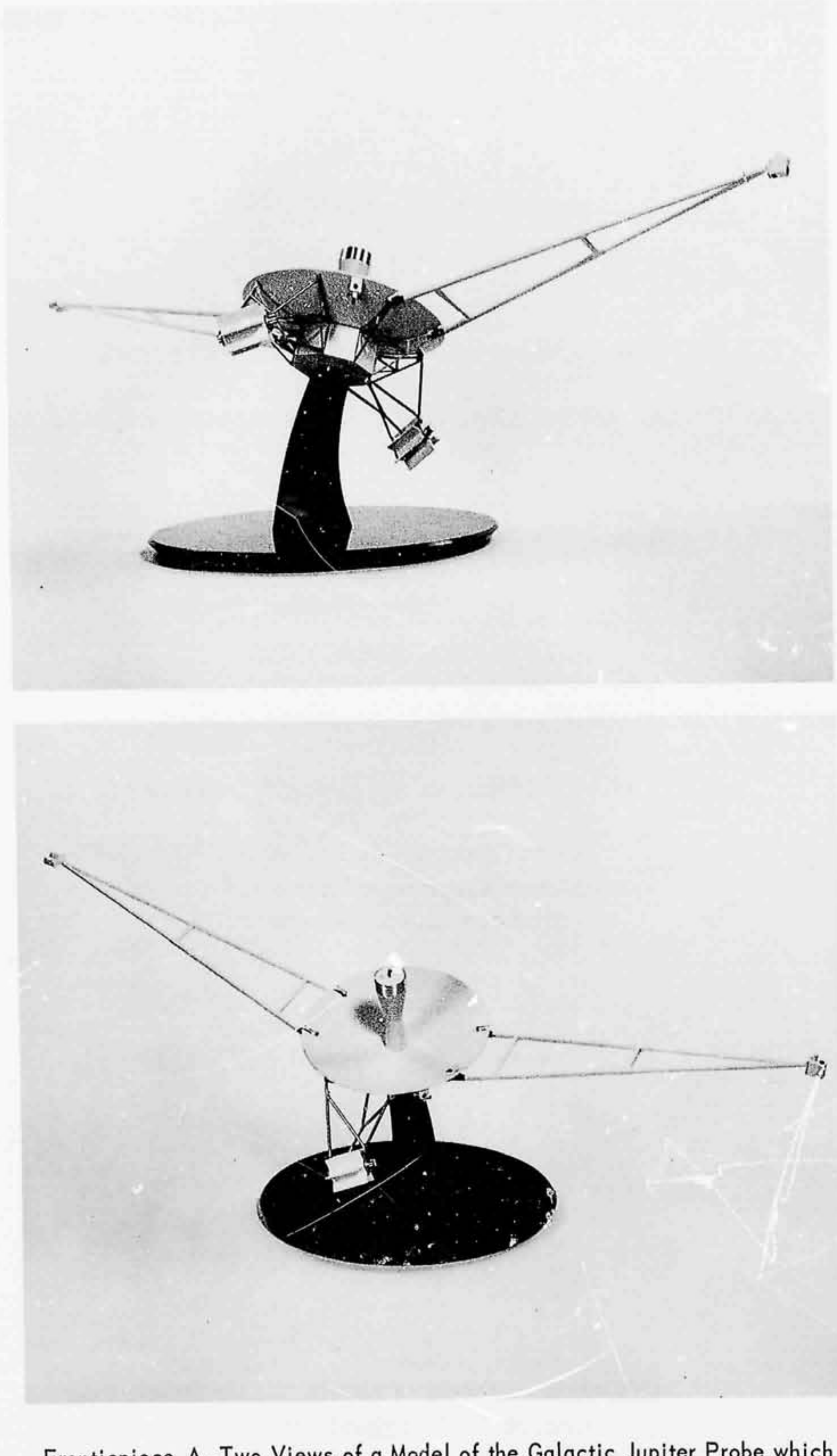
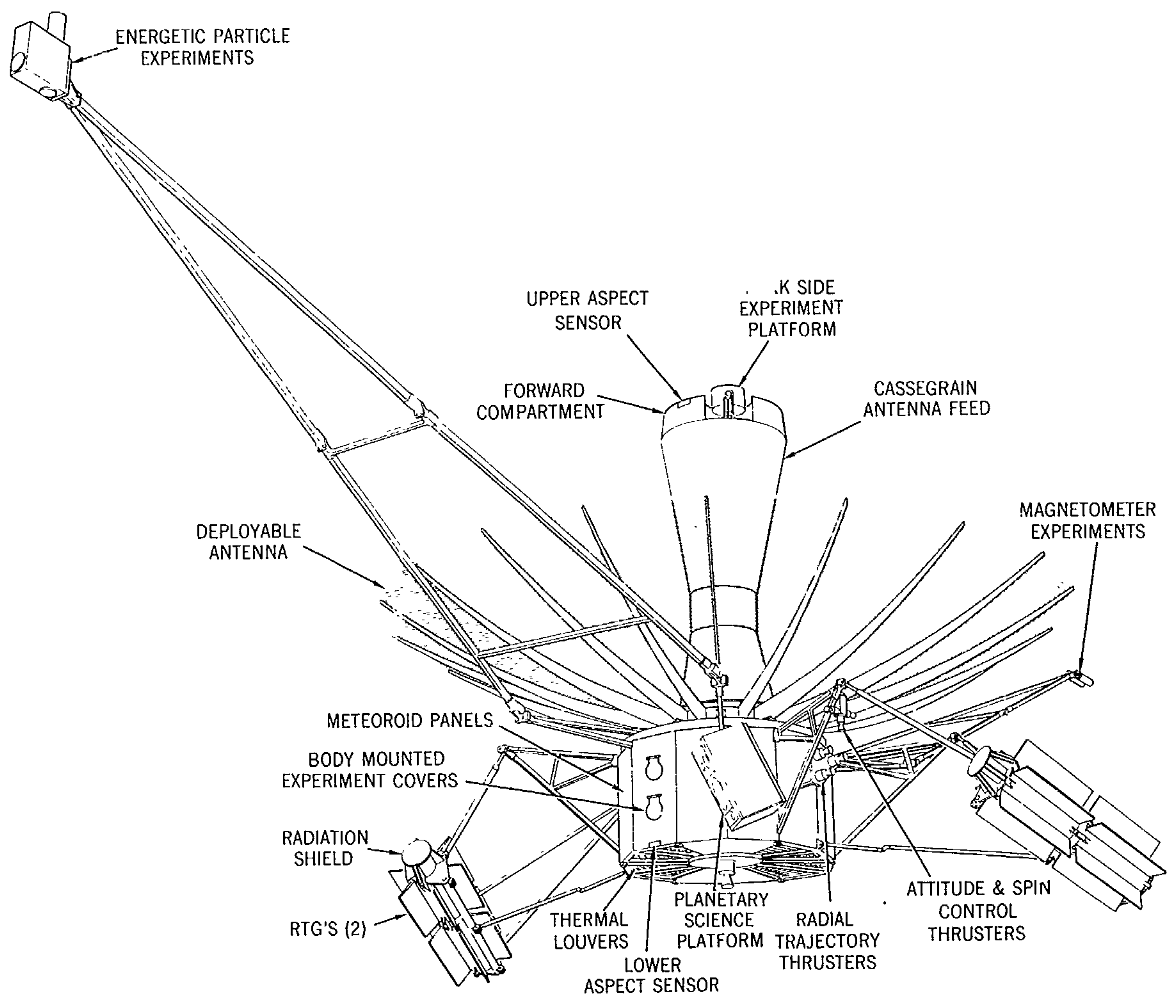
GODDARD SPACE FLIGHT CENTER
Greenbelt, Maryland 70771



Frontispiece A—Two Views of a Model of the Galactic Jupiter Probe which served as the "Reference Design" for this Task



Frontispiece B–Outer Planets Explorer

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ANALYSIS OF SELECTED

DEEP-SPACE MISSIONS

A Report Covering Task I Effort Under The Study

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ABSTRACT

This report covers NEW MOONS* study Task I, Analysis of Selected Deep-Space Missions and includes an introduction to considerations of launch vehicles, spacecraft, spacecraft subsystems, and scientific objectives associated with precursory unmanned missions to Jupiter thence out-of-the ecliptic plane as well as other missions to Jupiter and other outer planets. Necessity for nuclear power systems is indicated. Trajectories are developed using patched conic and n-body computer-techniques.

*NASA Evaluation With Models Of Optimized Nuclear Spacecraft (NEW MOONS) Contract NAS 5-10441, performed by RCA Astro-Electronics Division, Defense Electronic Products, Princeton, New Jersey for NASA Goddard Space Flight Center, Greenbelt, Maryland.

ACKNOWLEDGMENT

Analysis of Selected Deep-Space Mission

Task I

PROGRAM:

In the course of conducting the studies of the NEW MOONS program valuable assistance has been provided by many people representing various organizations. It is considered appropriate to identify those whose contributions were most vital.

Fred Schulman, NASA Office of Advanced Research and Technology and Marcel Aucremanne, NASA Office of Space Science and Applications both realized the necessity for the NEW MOONS studies and provided technical guidance and financial support throughout the program.

Daniel G. Mazur, Assistant Director for Technology and Rudolph A. Stampfl, Deputy Assistant Director both of Goddard Space Flight Center aided in program initiation.

RCA Astro-Electronics Division, the prime contractor, recognized the importance of the NEW MOONS program and has given its support and cooperation toward realizing the objectives of the program. Herbert Bilsky served in the capacity of RCA project manager and technically contributed to the program as well as provided aid in preparation and review of this report.

REPORT PREPARATION:

For this report J. Michael L. Holman of RCA AED was the principal investigator and author of the draft materials. Brian Stockwell of RCA AED, P. Michael Lion and J. Preston Layton of the Aerospace Systems and Mission Analysis (ASMAR) group of Princeton University and Sam Pines of Analytic Mechanics Associates provided valuable assistance to the prime contractor.

Rudolph A. Stampfl focused the effort in this task to include special emphasis on the out-of-the-ecliptic plane mission. Other Goddard Space Flight Center

personnel who have provided important technical information, review and comments include James Trainor, Emil W. Hymowitz, R. E. Coady, R. T. Groves, Joseph H. Conn, and William G. Stroud. George M. Levin in addition to reviewing and commenting on the complete report, also provided an appendix on launch vehicle considerations.

William S. West, Technical Officer
Goddard Space Flight Center
Greenbelt, Maryland

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PREFACE

BACKGROUND AND RELATED INFORMATION

Since the early 1960's, personnel of the Goddard Space Flight Center have been interested in deep-space missions to obtain information concerning the planets, Jupiter, Saturn, Uranus, Neptune and Pluto, as well as information concerning the interplanetary medium. Studies have been performed to establish the feasibility of such missions and various reports were written by Goddard personnel[1] and by others[2].

For almost as long as these missions have been considered, the engineers, scientists and managers at Goddard have realized the necessity for systems, independent of the Sun's energy, to supply the spacecraft electric power requirement. In general, Goddard studies have indicated that there is a weight advantage in using small nuclear power systems such as radioisotope fueled thermoelectric generators instead of presently available solar cells when missions go beyond 2.5 or 3 AU[3]. Further, there are technological and practical uncertainties in projecting use of solar arrays in a range starting beyond 3-5 AU[4] whereas the use of small nuclear power supplies is technically and practically feasible. However, the use of small nuclear systems, while feasible, nevertheless presents technical questions. An in-house Goddard study[5] identified pertinent technological areas requiring study prior to the use of these nuclear generators on spacecraft designed for scientific deep space missions[6]. These areas were divided into the following numbered tasks:

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[1] A selected list of Goddard Space Flight Center deep space reports includes the following references: 1, 2, 16, 25, 26, 27, 28, 29, 30, 31.

[2] A limited list of deep space reports prepared by other centers and contractors includes: 4, 32, 33, 34, 35, 36.

[3] See Reference 2, 28.

[4] Technical uncertainties involve practical design questions arising from the use of very large solar array areas, their survival through meteroid belts and their system performance when operating at the low temperature and low illumination levels anticipated. This topic is discussed in References 1 and 2.

[5] See Reference 28.

[6] This study is referred to as NEW MOONS.

Task Number	Task Description — Title	Reference X Document
I	Analysis of Selected Deep-Space Missions	X-701-69-170
IIA	Subsystem Radiation Susceptibility Analysis of Deep-Space Missions	X-701-69-171
IIB	Spacecraft Charge Build-Up Analysis	X-701-69-172
III	Techniques for Achieving Magnetic Cleanliness	X-701-69-173
IV	Weight Minimization Analysis	X-701-69-174
V	Spacecraft Analysis and Design	X-701-69-175
VI	Spacecraft Test Documentation	X-701-69-176
VIIA	Planar RTG-Component Feasibility Study	X-701-69-177
VIIB	Planar RTG-Spacecraft Feasibility Study	X-701-69-178
VIII	RTG Interface Specification	X-701-69-179
	Summary Report of NEW MOONS	X-701-69-190

Specific Rationale for Task I. Prior to conducting the NEW MOONS study, an analysis[7] of the OSSA 1964 and 1965 prospectuses was performed to determine which contemplated missions might require small nuclear power systems. Each prospectus indicated several (approximately 10 to 20) nuclear candidate missions. Initially, therefore, Task I was planned to focus more detailed engineering analysis to "confirm the necessity" for such nuclear power sources for at least certain missions. The deep space missions appeared to be the most likely missions to require nuclear power and accordingly Task I was limited to a consideration of out-of-the-ecliptic flights and one and two planet fly-by missions. A contract[8] was established for further study of these areas. This study was entitled NASA Evaluation With Models Of Optimized Nuclear Spacecraft (NEW MOONS). During the execution of the NEW MOONS Technology Study, Goddard

[7]See Reference 28

[8]NAS-5-10441 RCA Astro-Electronics Division, Princeton, N. J.

was assigned the task of conducting a Phase A study covering a Galactic Jupiter Probe[9]. These two study efforts, Galactic Jupiter Probe and NEW MOONS, were directed to provide the maximum practical benefit to each other. In general, the Galactic Jupiter Probe was considered as a "base line spacecraft and mission" or a "reference design" during the NEW MOONS Technology Study. On the other hand, the Galactic Jupiter Probe Study team made use of the technology and data as developed by the NEW MOONS Study in areas of missions analysis, shielding, aerospace nuclear safety, thermal and structural analysis and other related areas.

As the NEW MOONS contract was being concluded, the scope of Galactic Jupiter Probe project was broadened and adopted the name Outer Planets Explorer (OPE)[10]. The Outer Planet Explorer is considered for a generally more ambitious program than the original Galactic Jupiter Probe, in that the OPE is intended for a family of single and multiple planet missions.[11] This was considered and encouraged during the NEW MOONS Task I and provides some of the basic data for the program expansion from the GJP to OPE concepts. Also, Task I of NEW MOONS emphasizes various aspects associated with missions out-of-the-ecliptic plane. Further, in Task I, Goddard directed preliminary attention to an imaging system and although GJP did not provide for such a system, the OPE study presently includes such systems. An additional effort was added to NEW MOONS to define a stable platform to facilitate planetary imaging on a spin stabilized spacecraft.[12] Similarly, additional work is being directed toward imaging considerations at Saturn, Uranus, Neptune and Pluto.

The OPE, as presently visualized, encompasses spacecraft in the 1100-1400 pound class whereas the GJP "reference design-spacecraft" for the NEW MOONS Study was 500-600 pounds. This is a significant practical difference from a flight project viewpoint; however, the technology and techniques of NEW MOONS are generally applicable. Specific numeric values will be different when solutions are developed, but the techniques and rationale indicated in the NEW MOONS reports are applicable to the general problem of integrating and using small nuclear power systems on a scientific spacecraft designed for deep space missions.

[9]See References 1 and 2 and Frontispiece A.

[10]See Reference 37 and Frontispiece B.

[11]See Appendix IV for A Strategy For Exploration of the Outer Planets using a 750 and a 1000 pound class spacecraft.

[12]This is covered in Reference 37.

APPLICABILITY TO OTHER PROGRAMS

The NEW MOONS technology and techniques reported may have applicability or some relevancy to additional space missions that may in the future use nuclear systems such as planetary landers and rovers as well as applications spacecraft.

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A Report Covering Task I Effort Under The Study

NASA Evaluation With Models Of Optimized Nuclear Spacecraft
(NEW MOONS)

SECTION I

INTRODUCTION AND SUMMARY

A. INTRODUCTION

This report summarizes the results of Task I, "Analysis of Selected Deep-Space Missions," performed as part of the NEW MOONS Study.

The objective of Task I was to analyze several specific missions that would then be evaluated parametrically during the remaining NEW MOONS Program effort. The analysis included the following specific Model Missions, the first three of which were outlined at the inception of the study and the last which was an outgrowth of the work performed in Task I:

(1) Jupiter-swingby out-of-ecliptic, 1972 launch

(2) Jupiter-swingby out-of-ecliptic, 1974 launch

(3) Grand Tour of the solar system, including multiple-planet swingby operations

(4) Two-planet swingby, such as Earth-Jupiter-Saturn, Earth-Jupiter-Uranus, Earth-Jupiter-Neptune

All of these missions were examined, in varying degrees of detail, using the Goddard Space Flight Center Phase-A Galactic Jupiter Probe (GJP) Study (Ref. 1) as a baseline spacecraft concept and considering existing launch vehicles or variants thereof.

Primary emphasis was directed toward the out-of-ecliptic Jupiter swingby missions because they provide a logical next step in increasing complexity over the baseline in-ecliptic mission, which was emphasized by GSFC in the Phase-A Study. The out-of-ecliptic missions impose more severe guidance requirements than the in-ecliptic mission, but these can readily be accommodated within the basic GJP capabilities. A Jupiter-swingby mode provides a latitude profile of scientific information for a lower expenditure of launch energy than a direct out-of-the-ecliptic launch from Earth. In addition, these missions would serve as valuable precursors to the more ambitious multiple-planet swingbys which would follow. The technology developed and exercised for the out-of-ecliptic mission and the better information which would thereby be provided on the interplanetary and Jovian environment will increase the probability of successfully achieving the objectives of the later multiple-planet flights.

The Grand Tour missions, consisting of sequential flybys of Jupiter, Saturn, Uranus, and Neptune impose severe guidance requirements which make systems based on Earth-based tracking alone inappropriate. Communications, data handling, power, and lifetime requirements all appear formidable. These and other factors were examined in this study, and where deficiencies were found to exist they are identified as items requiring further study. The level of enhancement required in several subsystem areas over the baseline GJP capabilities would inevitably require considerable early investment to prepare for a launch in the 1977 or 1978 opportunities.

As a result of the Grand Tour evaluation, consideration was subsequently given to two-planet swingbys, which emerged as logical extensions of the out-of-ecliptic missions. Preliminary analysis indicated that Jupiter swingbys to Saturn, to Uranus, and to Neptune could be accomplished with minimum modification and growth of the baseline GJP. Using Saturn as the first planet for swingbys to Uranus and Neptune extends the launch opportunities into the 1980's. Exploration of the outer planets can therefore proceed in an orderly manner using a spacecraft with gradually increasing capabilities which can evolve from the baseline GJP vehicle.

Evaluation of the Model Missions included analysis of ballistic trajectory parameters including launch opportunities, required injection energies and launch-vehicle capabilities. The scientific objectives of the out-of-ecliptic missions cover investigation of the physics of both interplanetary space and the planetary environment, including the measurement of particle radiation and magnetic fields in both environments and the temperature and pressure distribution within the planetary atmosphere. These objectives were evaluated in terms of a set of experiments that would be appropriate to the missions being considered, and which can be supported by the GJP. Subsequent planetary flybys will have similar objectives but in the case of the Grand Tour possibly more limited capabilities.

Based on the mission requirements and scientific objectives, subsystem functional requirements were examined for the RTG power supply, both in terms of performance and impact on other subsystems and in the areas of attitude control, data handling, trajectory correction capability, thermal control and communications. The subsystem requirements were then compared to the baseline GJP capabilities and conclusions on their suitability for the more advanced missions are presented. The Task I mission analysis also provided data to other tasks of the NEW MOONS Program, specifically, to Task II-A, " Subsystem Radiation Susceptibility Analysis;" Task II-B, "Spacecraft Charge Buildup Analysis;" Task IV, "Weight Minimization Analysis;' and Task VII-B, "Spacecraft-Planar RTG Feasibility."

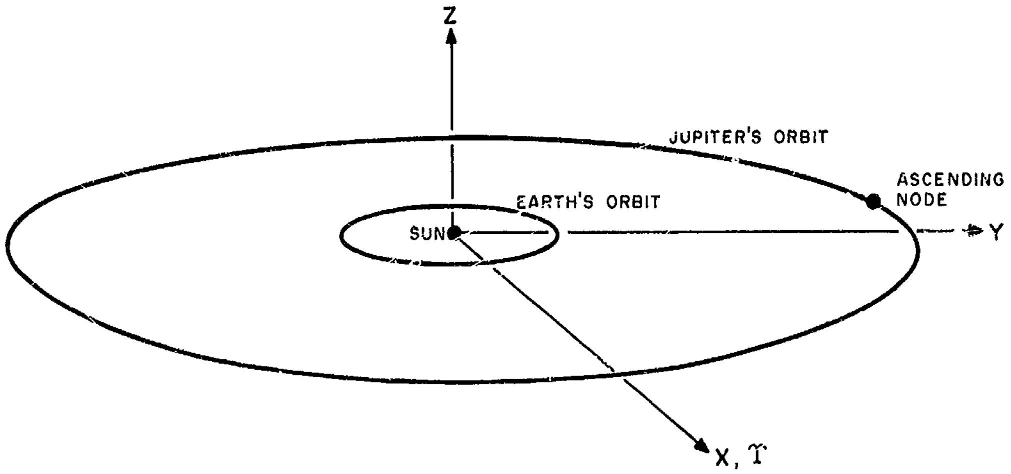
B. SUMMARY

1. Mission Descriptions

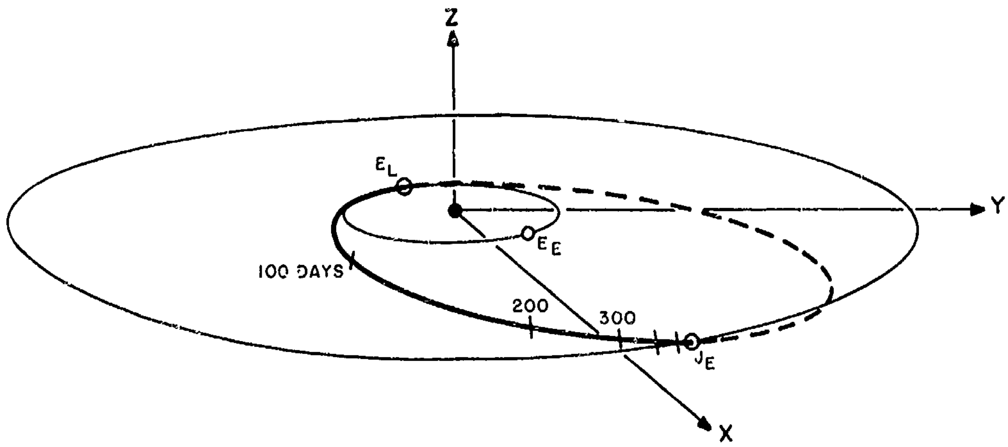
The out-of-the-ecliptic missions, use the gravitational field of Jupiter to deflect the heliocentric orbit of the spacecraft significantly out of the ecliptic plane to provide a latitude profile of scientific measurements for comparison with the presently available in-ecliptic measurements.

An overall view of an out-of-the-ecliptic trajectory is shown in Figure 1. The orbit of Earth defines the ecliptic plane (View A of Figure 1) with a Sun-centered axis system such that X and Y lie in the ecliptic and Z normal to it. Jupiter's orbit lies close to the ecliptic, at an inclination of approximately 1.3°. During its journey from Earth to the vicinity of Jupiter the probe also remains very close to the ecliptic as shown in View B of Figure 1. Tick marks at one-hundred-day intervals indicate the spacecraft's progress from Earth at launch (E_L) to its encounter with Jupiter (J_E) approximately 550 days later. It is evident that at encounter the communication distance to Earth (J_E to E_E) is near a minimum for such a transfer. If the spacecraft trajectory were not perturbed by Jupiter, the probe would cross Jupiter's orbit and continue indefinitely in its near-ecliptic elliptical orbit of the Sun. However, by carefully selecting the aiming point at Jupiter the spacecraft orbit can be deflected out of the ecliptic, as shown in View C of Figure 1, so that by 600 to 700 days from launch the probe reaches an appreciable distance above the plane and continues to a maximum elevation of more than 1 AU at approximately 1050 days from launch. Such trajectories satisfy most of the requirements for out-of-ecliptic scientific observation and are much more economical in terms of launch energy than an orbit inclined at 90° to the ecliptic plane.

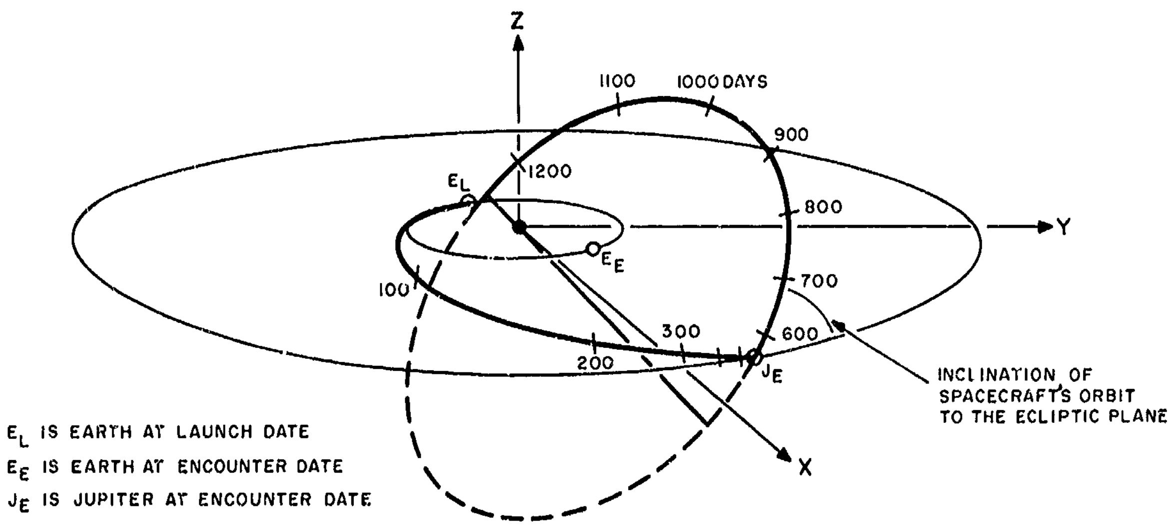
The third mission, the Grand Tour, is defined to be a sequential flyby of the planets Jupiter, Saturn, Uranus, and Neptune at sufficiently small distance of closest approach to allow meaninfgul scientific observation of each of the planetary



A. ORBITS OF EARTH AND JUPITER



B. EARTH-JUPITER TRAJECTORY



C. POST-ENCOUNTER TRAJECTORY

Figure 1. Out-of-Ecliptic Trajectory

environments.* During the period 1976 to 1980, annual opportunities to perform such missions exist, at least in theory, and since these opportunities do not recur until 2154 AD, there is considerable interest in examining their requirements at this time.

The two-planet swingby, which was not one of the Model Missions set forth in the governing work statement, represents a less-ambitious scheme than the Grand Tour, but a very logical follow-on to the out-of-ecliptic missions. Secondary-target planets leading to missions such as Earth-Jupiter-Saturn, Earth-Jupiter-Uranus, or Earth-Jupiter-Neptune were considered, though not to the level of detail of the out-of-ecliptic missions. Opportunities for Jupiter-swingby missions to the outer planets occur in the 1976 to 1980 period, with overall flight times of the order of 3 years to Saturn, 6 years to Uranus, and 9 years to Neptune. Saturn swingbys to Uranus and Neptune will be possible during the 1980's.

2. RTG Power Supply

The selection of a power-supply system for a particular mission is based on the environment in which the probe is to operate, its operational lifetime, and the power level needed. Comparing various power sources at a 100-watt(e) level for deep-space missions with lifetimes of three years or more it has been shown (Ref. 2) that neither batteries, fuel cells nor reactors are competitive with conventional solar-cell systems or RTG sources. It has also been shown that the weight of a conventional N/P silicon solar-cell system exceeds that of an RTG at a solar range of approximately 2.7 AU. For an ideal silicon solar-cell system, the crossover occurs at about 3.2 AU. Advanced thin-film solar-cell technology shows promise of matching the RTG weight to a range of 5 AU, but the array area needed at such ranges becomes very large, of the order of 400 ft^2. For missions to Jupiter at a mean solar distance of 5.2 AU and beyond, an RTG power system appears to be the most reasonable choice in the low-power range.

3. Trajectory Analysis

A trajectory analysis has been performed for the three Model Missions and the subsystem requirements that were developed (and also projected to a two-planet swingby) have been compared with the capabilities of the Galactic Jupiter Probe (GJP) concept, described in Reference 1.

The basic GJP spacecraft, weighing between 550 and 600 pounds, can be launched by a SLV3C/Centaur/TE 364-4 launch vehicle on a fast, nominally

*For further definition of Grand Tour see Section II-C. At time of this writing interest continues in the Grand Tour (see Ref. 38).

550-day, flight to Jupiter during both the 1972 and 1974 launch opportunities. These trajectories lead to near-minimum communication ranges (6×10^8 km) at Jupiter encounter and approach velocities which are capable of yielding significant post-encounter inclinations to the ecliptic ($> 50°$ in 1972 and $> 40°$ in 1974). A scientifically useful out-of-the-ecliptic mission should be capable of gathering data at a distance of approximately 1 AU above the ecliptic plane in the vicinity of Earth's orbit, and this can be achieved at both launch opportunities.

The effect of aiming-point variation at Jupiter has been investigated and suitable aiming zones identified such that

(1) The spacecraft flies sufficiently close to the planet (8 to 10 planetary radii) to perform significant encounter measurements,

(2) The post-encounter trajectory reaches a distance of more than 1 AU above the ecliptic plane in the neighborhood of Earth's orbit, and

(3) The perihelion distance of the spacecraft is greater than 1 AU during the post-encounter phase so that the variation in solar input does not impose intolerable demands on the thermal-control system.

Launch-injection errors can be reduced by a single arbitrary-pointing midcourse correction maneuver ($\Delta V < 100$ m/sec) to a circle radius 75,000 km (3σ) at Jupiter encounter, which is compatible with the aiming zones identified above. After encounter, the spacecraft climbs out of the ecliptic plane, reaching a maximum distance of 1.2 AU after 1050 days, subsequently spending about 200 days at more than 1 AU above the plane. During this period of maximum scientific interest, the spacecraft-Earth distance is in the range of 2.5 to 3.3 AU, so that the communication capability is considerably higher than that available at Jupiter encounter (~800 bps). For the out-of-the-ecliptic missions, the spacecraft-Sun distance is a maximum at encounter so that the variation in solar input is considerably less than that experienced in the baseline mission.

A set of objectives for the Grand Tour mission (Table 1) are discussed involving closest approaches to Jupiter, Saturn, Uranus, and Neptune, of the order of ten planetary radii, and the possibility of implementing such missions briefly examined. For reasonable values of launch energy, a 1976 flight requires a very severe deflection angle at Jupiter to reach Saturn, which can only be achieved by an excessively close flyby. In 1980 the deflection at Jupiter is minimal and, correspondingly, the flyby distance very large. Between the opening and closing of this series of opportunities, intermediate flights in 1977, 1978, and 1979 appear satisfactory with overall flight times of the order of 10 years to Neptune at a solar distance of approximately 30 AU. The guidance accuracy requirements for even a two-planet swingby are an order of magnitude more severe than those for an out-of-the-ecliptic mission, and the progressive accumulation of errors in sequential flybys requires additional analysis to demonstrate the feasibility of a Grand Tour.

Table 1

Grand-Tour Trajectory Objectives

Earth-Jupiter interplanetary science	1 to 5 AU
Jupiter-Encounter science	5 to 10 R_j
Jupiter-Saturn interplanetary science	5 to 9 AU
Saturn-Encounter science	< 1.2 or > 2.5 R_s
Saturn-Uranus interplanetary science	9 to 18.5 AU
Uranus-Encounter science	2.5 to 5 R_u
Uranus-Neptune interplanetary science	18.5 to 30 AU
Neptune encounter	< 10 R_n

For the case of the two-planet swingby, for example, an Earth-Jupiter-Saturn Mission, the spacecraft requirements are not greatly different from those for an out-of-the-ecliptic mission. The principal change required is again in the area of guidance. In addition to an arbitrary-pointing mid-course correction of up to 100 m/sec, applied within ten days of launch, a second pre-encounter correction of up to 5 m/sec after about 100 days of Earth-based tracking is indicated. The second correction reduces the aiming-point errors at Jupiter to tracking residual and ephemeris errors, approximately a 5000-km-radius circle about the nominal aiming point. If these errors were allowed to propagate over the Jupiter to Saturn leg of the trajectory, the uncertainty in passage distance at Saturn would be some tens of planetary radii which is hardly adequate for scientific investigation of the planetary environment. The spacecraft velocity error at Jupiter departure could be determined from Earth-based tracking and a post-encounter correction, of the order of 50 m/sec, should reduce the uncertainty in Saturn fly-by distance to approximately one-half a planetary radius. This would allow an exterior ring passage of Saturn with a nominal passage distance of say five radii, sufficiently close to allow good scientific observation without endangering the spacecraft. The overall duration of such a mission from Earth launch to Saturn arrival would be from 3 to 4 years and the launch-energy requirements in the range of $C_3 = 90$ to $120\ km^2/sec^2$, depending on year of launch.

For a trailing edge swingby of Saturn, the probe continues in its hyperbolic orbit to cross the orbits of Uranus and Neptune, though the limited flyby accuracy at Saturn is not sufficient to obtain flybys of the outer planets. The communication capability of the basic GJP spacecraft has been designed with a 10 AU mission in mind. At Saturn encounter the 9-ft dish and 10-watt transmitter provides more than 100 bps into a 210-ft antenna at Earth, and 10 bps to beyond 20 AU.

Generally then, the basic GJP capabilities with provision of approximately 1.5 times the nominal midcourse ΔV capacity and an arbitrary pointing capability would be capable of performing an EARTH-Jupiter-Saturn swingby.

4. Assessment of Launch-Vehicle Capabilities

The Atlas SLV3C/Centaur/TE 364-4 has in earlier paragraphs been identified as suitable for use for the out-of-ecliptic missions and is also appropriate to two planet swingbys using Jupiter assist. However, with the larger launch energy requirements of the later opportunities and to provide for an increase in spacecraft weight, the desirability for SLV3X first-stage booster is indicated.* The increased payload capabilities of the SLV3X or Titan IIID first-stage boosters permits the use of a final stage which is guided through injection in place of the spinning TE 364-4. The aiming-point error ellipse at Jupiter corresponding to the SLV3X/Centaur/Burner II or the Titan IIID/Centaur is much smaller than that due to the SLV3C/Centaur/TE 364-4. Work performed at GSFC since the completion of this study indicates that the injection errors are sufficiently small to reduce the on-board trajectory-correction requirements to the order of 30 meters per second, well within the capabilities of the basic GJP spacecraft.

Since the Titan IIID/Centaur is capable of supporting payloads in excess of 1200 pounds for a characteristic velocity of 49,000 ft/sec., it becomes a likely candidate for the Grand-Tour Mission, which will probably require a much larger spacecraft than the baseline GJP.

5. Scientific Objectives

The scientific objectives of all four postulated missions are essentially similar. Interplanetary particle and field measurements will be made during

*See Section III. Also see Appendix 3 for discussion of the performance of alternate candidate launch vehicles. See Section III for comments concerning availability of the SLV3X.

the cruise phases of the missions to extend their spatial coverage and to attempt to define the limits of the organized solar wind. During planetary encounters, scientific investigations will include measurements of the magnitude of magnetic fields and the density of trapped radiation belts, as well as remote soundings of the planetary atmosphere and surface. A representative scientific payload has been chosen to exercise the spacecraft design in terms of power, size, and weight allocations and to generate a typical profile of scientific data. The experiment list contains a sensitive magnetometer and high-energy particle detectors; consequently, radiation and magnetic fields produced by the RTG imposes constraints on spacecraft design in order to minimize the background noise.

Because of the general interest in obtaining a close up view of the planet, various imaging experiments were considered. A television.camera-magnetic tape recorder system was selected which is capable of providing an order of magnitude improvement in surface resolution compared with Earth-based photography and which is compatible with the spacecraft's spin stabilization and nuclear power source. The weight and power requirements of this imaging system, however, would severely restrict other desirable scientific measurements on a vehicl^ of the GJP class.

6. Conclusions

To conduct the wide range of deep-space missions studied in this Task, a power system independent of incident solar energy is a necessity. In the power range of 100 watts (e), RTG's are the most reasonable power source (Ref. 2).

Generally, it was found that the out-of-the-ecliptic mission requirements could be met by the GJP capabilities. In particular, the communication and thermal requirements are less severe than those of the 10 AU in-ecliptic mission. The principal change required in the baseline configuration is the provision of an alternative celestial reference system for closed-loop control of arbitrary pointing during the trajectory-correction maneuver and during cruise at a substantial angle out-of-the-ecliptic plane.

For the Grand Tour Mission, the present GJP capabilities require considerable upgrading. Estimates of the extent and methods of achieving this increased capability in, for example, the communication, data-storage, and thermal-control subsystems is relatively straightforward. In some areas, however, such as trajectory correction and on-board guidance, it is presently difficult to define the requirements due to uncertainty in planetary ephemerides, and orbit-determination accuracy.

The potential of the GJP to perform two-planet (e.g., Earth-Jupiter-Saturn) swingbys during the 1976-80 launch opportunities is sufficiently encouraging to warrant more detailed analysis than was possible within the scope of the present study. It appears that the principal change required in the baseline configuration is the provision of an increased midcourse ΔV capacity and an arbitrary-pointing capability for the Earth-Jupiter-Saturn Mission. The use of the spacecraft may be extended to cover such missions as Earth-Jupiter-Uranus, and Earth-Jupiter-Neptune flights in 1978-1982; and Earth-Saturn-Uranus or Earth-Saturn-Neptune flights through the 1980's.

SECTION II

TRAJECTORY ANALYSIS

A. CHOICE OF INTERPLANETARY TRAJECTORIES

An overall view of Earth-Jupiter ballistic trajectory parameters for the 1968-73 opportunities has been provided by Clarke (Ref. 3) and recently extended to the 1974-80 opportunities (Ref. 4). These data are obtained from a patched conic trajectory program, such as that described in Appendix I, where, at any instant, the spacecraft is regarded as being under the influence of a single attracting body, in this case, Earth, the Sun, and Jupiter, in turn.

The probe reaches the edge of Earth's sphere of influence (~144 Earth radii) approximately one day after launch. It then flies essentially under the influence of the Sun alone until it reaches Jupiter's sphere of influence (~675 Jupiter radii) approximately fifty days before closest approach. Patching together the hyperbolic Earth-departure trajectory, the elliptical heliocentric phase, and the hyperbolic Jupiter-encounter phase gives a very good approximation to the actual trajectory. Key parameters of the Earth-departure phase, the heliocentric-transfer phase, and the Jupiter-arrival phase are plotted on an arrival-date versus launch-date grid. These charts enable the most appropriate group of trajectories for a mission to be selected from the complete range of possible trajectories.

After a choice of trajectory has been made, the patched conic approximation can be replaced by an analysis which continually takes account of the influence of all solar-system bodies on the spacecraft orbit. During the present study, precision n-body trajectories were generated for the selected aiming zones at Jupiter, using a modified version of the ITEM program (Ref. 5). The modifications to the program are included as Appendix II. Generally the n-body results confirmed the patched conic results with only minor modifications to the out-of-the-ecliptic trajectory parameters.* The small differences were due primarily to the influence of the Sun during the one hundred days which the spacecraft spends inside Jupiter's sphere of influence.

Consider the launch-energy vs. time-of-flight contours for 1972 and 1974, shown in Figures 2 and 3, respectively. It can be seen that for both launch dates,

*The first successful n-body computer runs were completed in July 1967 which indicated an acceptable design for a trajectory covering a flight out-of-the-ecliptic plane. Somewhat prior to this, sufficient patched conic computer runs were completed to give substantial confidence in the techniques employed.

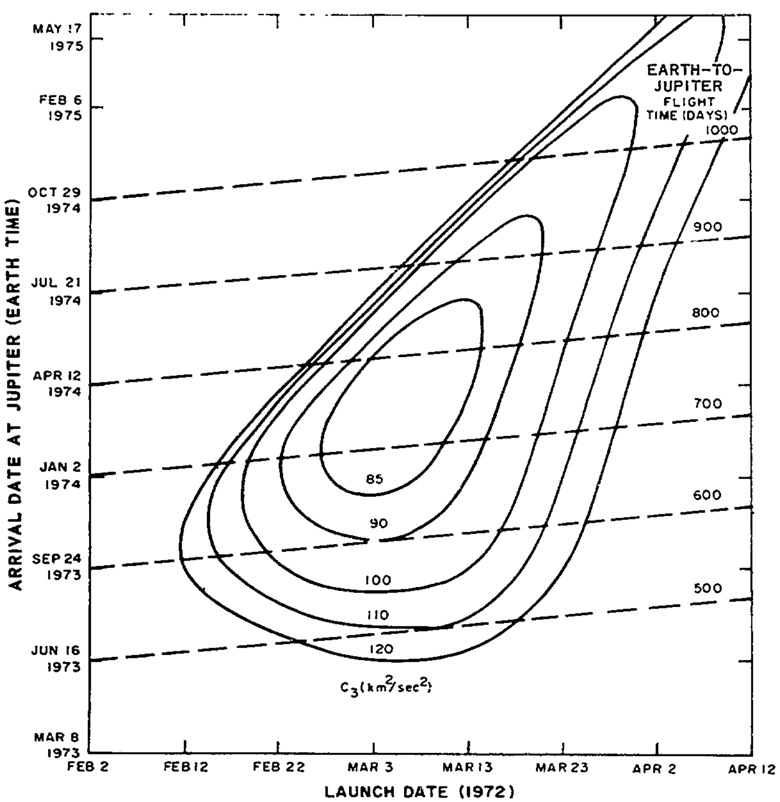


Figure 2. Launch Energy vs. Flight Time (1972)

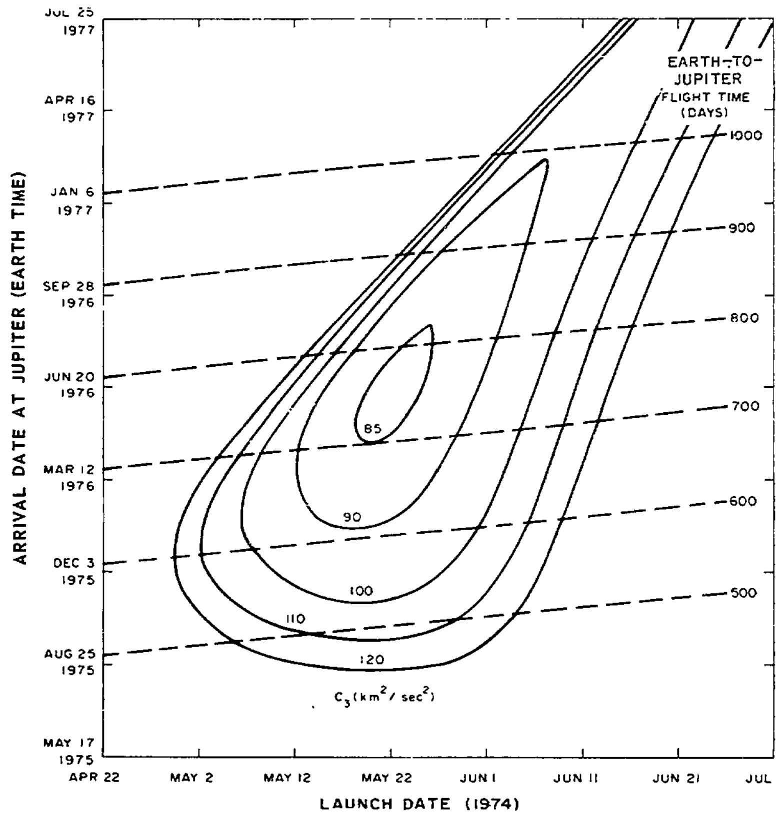


Figure 3. Launch Energy vs. Flight Time (1974)

minimum-energy trajectories, with $C_3 \leq 85$ km^2/sec^2, require flight times of approximately two years. In addition to the adverse effect on reliability of such long flight times, they also lead to maximum Earth-to-spacecraft communication distance at encounter.

Shorter flight times, of course imply higher launch energies and consequently lower spacecraft weight for a given launch vehicle. For a spacecraft of the Galactic Jupiter Probe class, with a weight budget of 550 to 600 lb, Earth-to-Jupiter transfers in the range of 500 to 600 days are achievable with variants of the Atlas-Centaur-Kick launch vehicles as discussed in Section III of this report. These shorter flight times lead to near-minimum communication distance at encounter (6×10^8 km compared to 9×10^8 km) and generally lead to Jupiter-approach parameters that are consistent with the post-encounter objectives of the model missions. In particular, the hyperbolic exceed speed (v_{hp}) of the Jupiter-approach hyperbola is in the range of 10 to 12 km/sec compared with ~7 km/sec for the slow trajectories, as shown in Figures 4 and 5. It is the approach velocity, together with the impact parameter $\bar{B}$, that control the hyperbolic flyby of the planet and the post-encounter heliocentric phase of the trajectory. High approach speeds are required to achieve significant post-encounter inclination to the ecliptic whereas the low approach speeds are more appropriate to establishing planetary orbits.

The angle between the approach asymptote and the Jupiter-Sun line (ζ_p) is typically of the order of 150° for the fast transfers, compared with 120° for the slow transfers, as shown in Figures 6 and 7, so that a better view of the sunlit face of the planet is given during approach on a fast trajectory.

The declination of the launch asymptotes (DLA) for the 500-to-600 day trajectories for both 1972 and 1974 lie in the range -20° to -30°, as shown in Figures 8 and 9. This range is within the limits of $-33.5° \leq \text{DLA} \leq 12°$, established in Reference 6 for 1-hour launch windows, 70° to 108° launch azimuth limits, and 25-minute Centaur coast between first and second burns. In addition, the spacecraft latitude during the first few days after launch is suitable for accurate trajectory determination from Earth-tracking data.

In summary, the most suitable interplanetary trajectories are those with flight times to Jupiter in the range of 500 to 600 days since they are (1) the fastest that can be achieved using the baseline launch vehicle, (2) lead to the most desirable Jupiter-encounter parameters, and (3) conform to launch-geometry constraints.

A more detailed view of the launch-energy requirements for such Earth-Jupiter transfers in the 1972 and 1974 opportunities are shown in Figures 10 and 11, respectively. These show similar Earth departure speeds ($v_h^2 = C_3$) for the

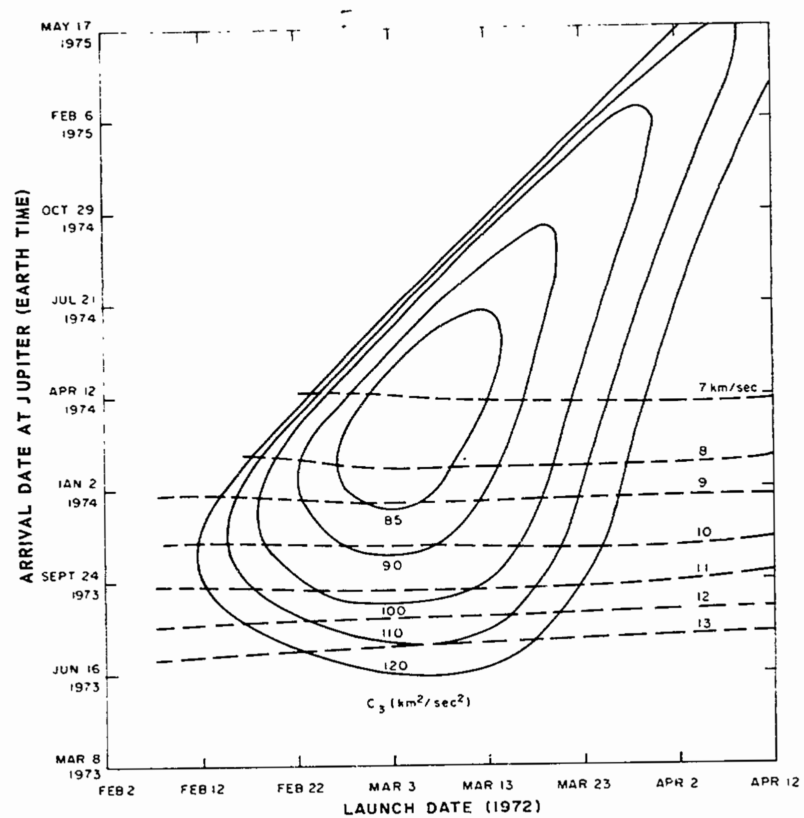


Figure 4. Asymptotic Approach Velocity (1972)

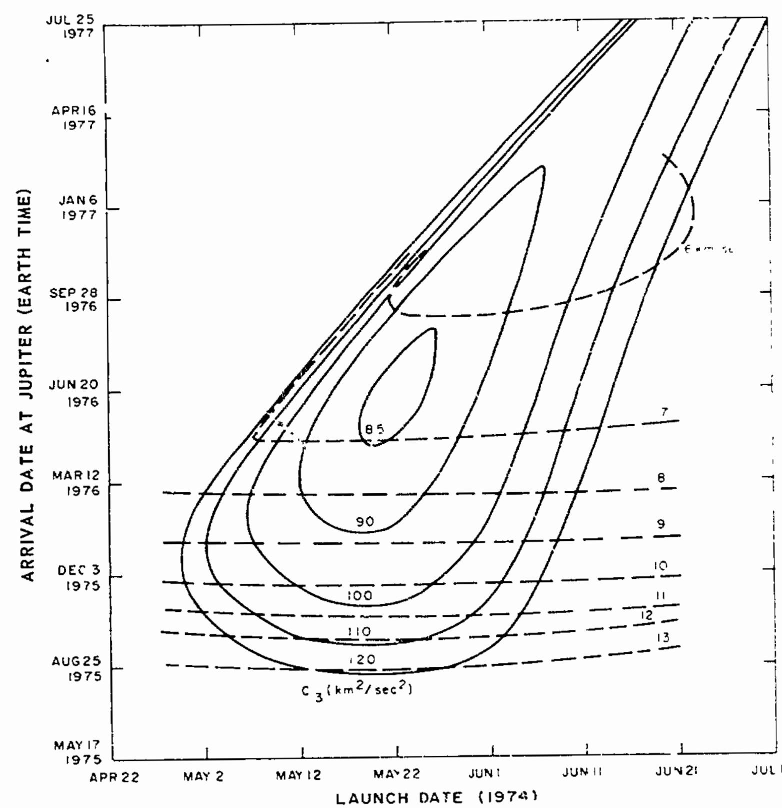


Figure 5. Asymptotic Approach Velocity (1974)

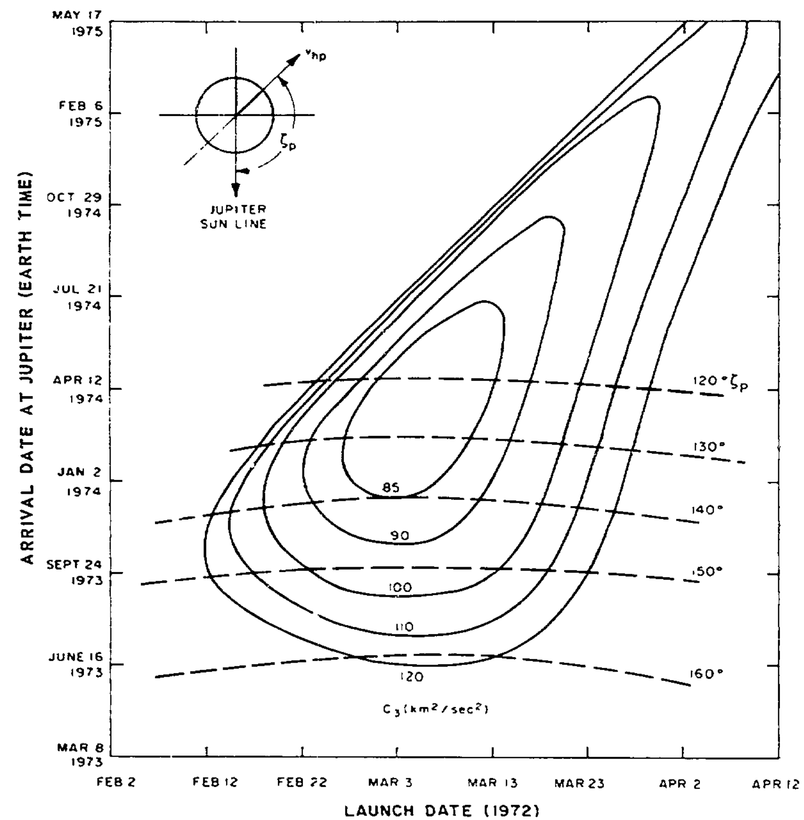


Figure 6. Angle Between Approach Asymptote and Jupiter-Sun Line (1972)

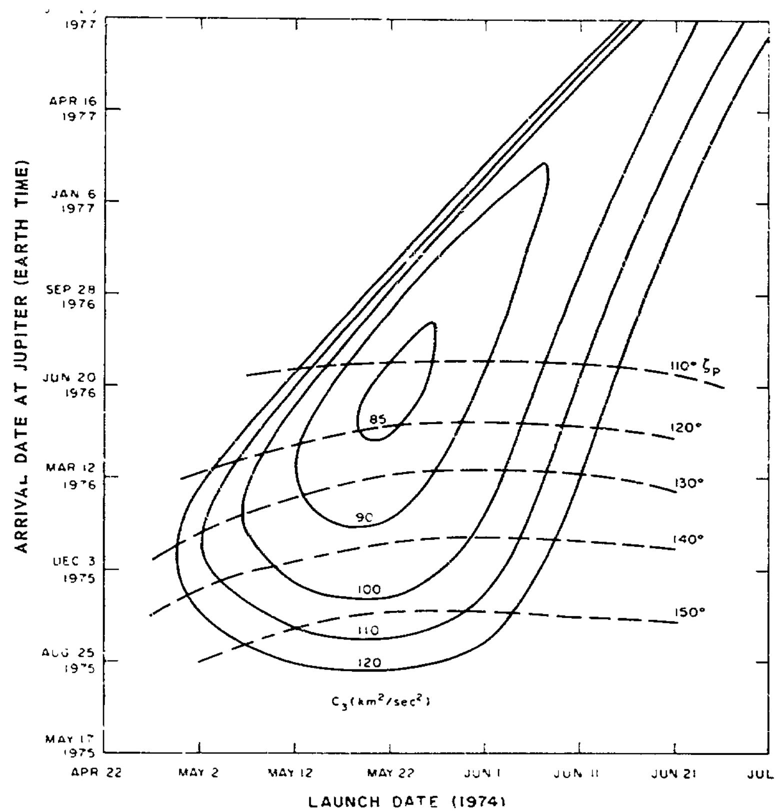


Figure 7. Angle Between Approach Asymptote and Jupiter-Sun Line (1974)

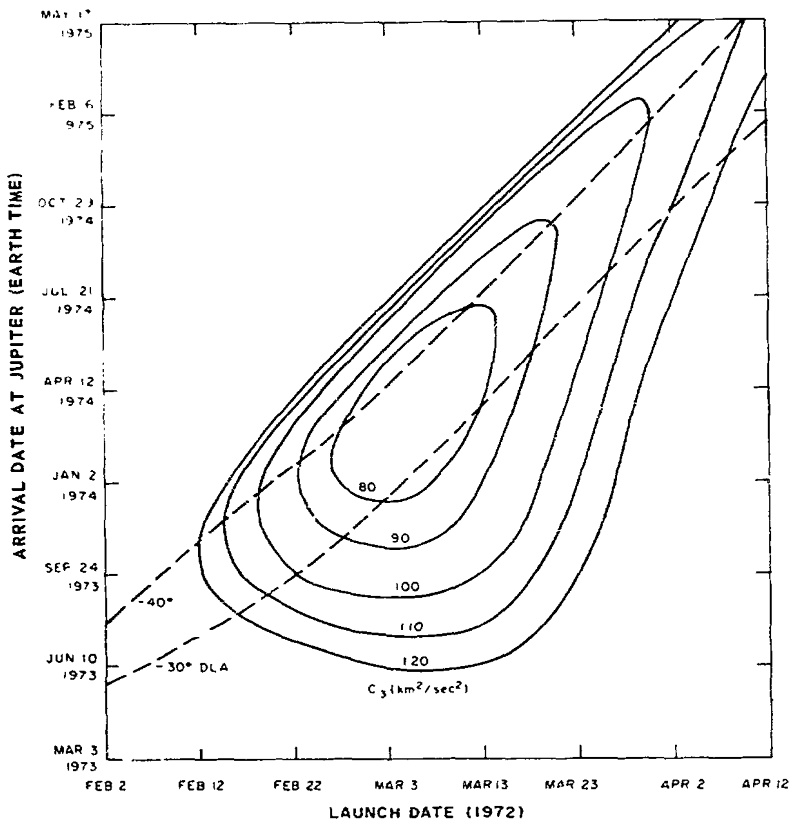


Figure 8. Declination of Launch Asymptote (1972)

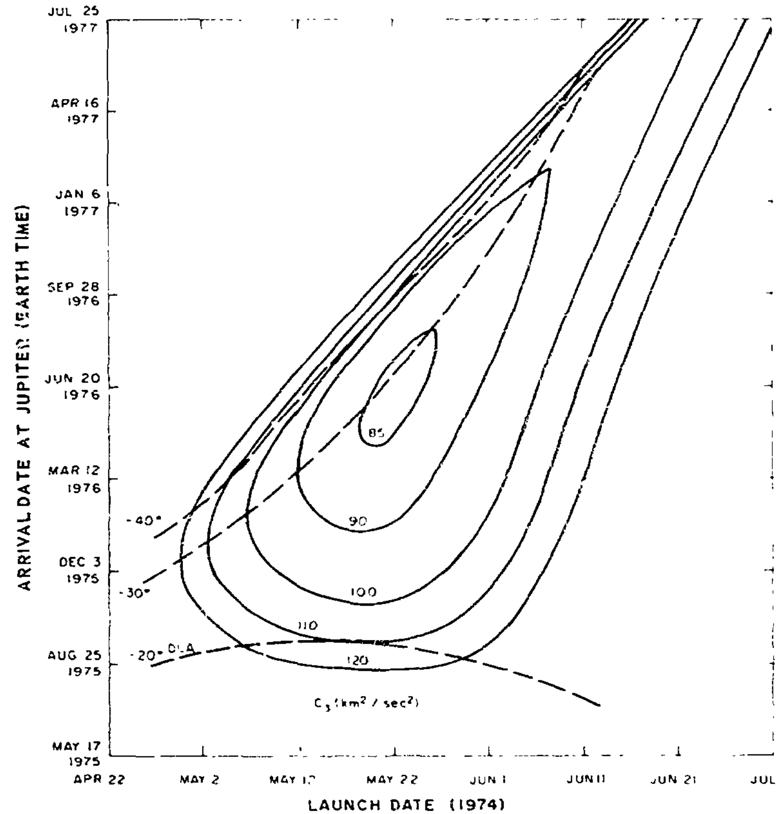


Figure 9. Declination of Launch Asymptote (1974)

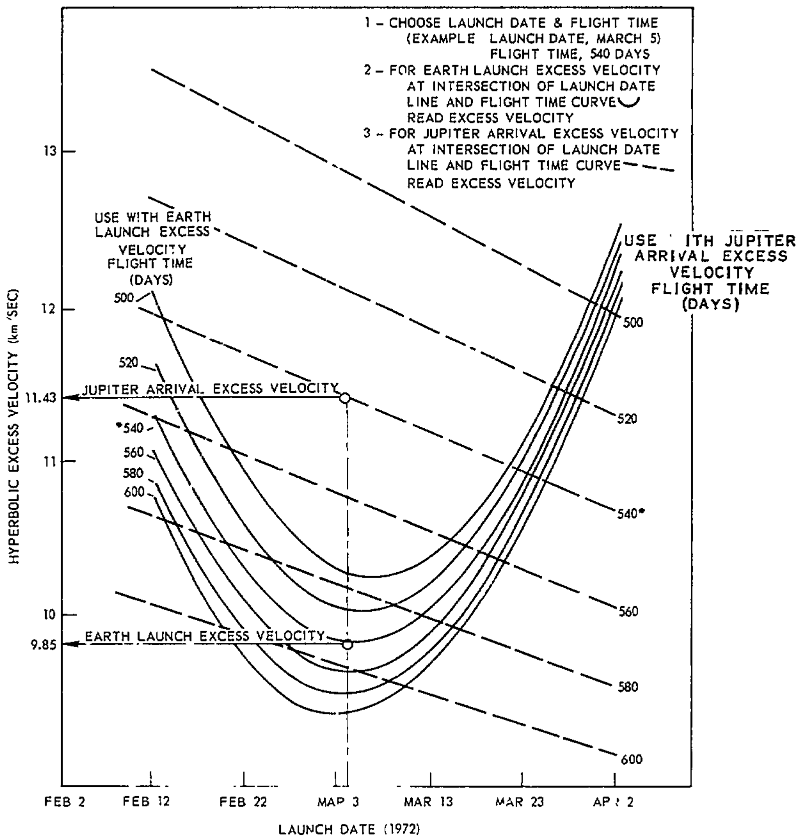


Figure 10. 1972 Launch Opportunity

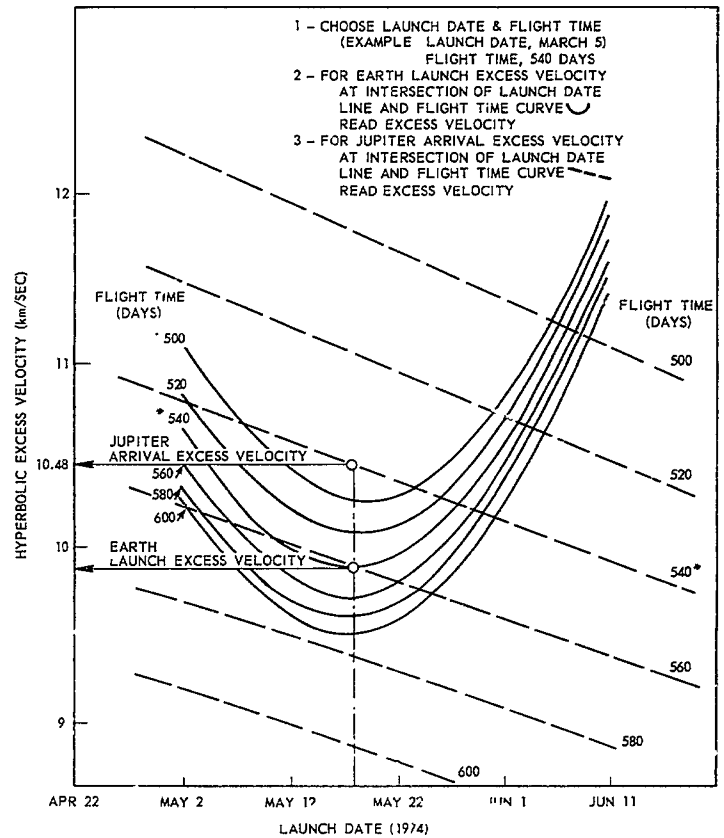


Figure 11. 1974 Launch Opportunity

two opportunities, for example, the single-opportunity minimum required for 540-day trajectories at either opportunity is ~ 9.85 km/sec ($C_3 \sim 97$ km^2/sec^2). However Jupiter-approach asymptotic speeds are consistently lower in 1974 than for the same flight durations in 1972, for example $v_{hp} = 10.48$ km/sec compared with 11.43 km/sec for the 540-day, minimum-energy launch dates. The post-encounter inclinations which can be achieved are closely related to the approach speeds. The maximum inclination for 550-day trajectories is approximately 55° in 1972 and 48° in 1974.

Precision n-body trajectories have been generated for 550-day Earth-Jupiter flight times with launch dates March 6, 1972 (J. D. 244.1383) and May 20, 1974 (J. D. 244.2188). Both dates are within suitable 20-day launch intervals such that the preferred launch vehicle, Atlas SLV3C/Centaur 70/TE 364-4 is capable of injecting the 550-lb GJP into the desired trajectory.

B. JUPITER-CENTERED AND POST-ENCOUNTER TRAJECTORIES

When the spacecraft is within about one-third of an AU of Jupiter, its trajectory within the so-called sphere of influence is accurately represented by the two-body equations appropriate to the initial, or entry, conditions with Jupiter as the sole central attracting body. The radius of the sphere of influence is given by

$$R_s = \left(\frac{m_{♃}}{m_{\odot}}\right)^{2/5} R_{\odot ♃} \tag{1}$$

where

$m_{♃}$ is mass of Jupiter,

$m_{\odot}$ is mass of the Sun, and

$R_{\odot ♃}$ is mean distance of Jupiter from the Sun.

The Jupiter approach velocity, that is the velocity of the spacecraft with respect to Jupiter at entry into Jupiter's sphere of influence, is given by

$$\bar{v}_1' = \bar{v}_1 - \bar{v}_p \tag{2}$$

where $\bar{v}_1$ is the spacecraft velocity with respect to the Sun and $\bar{v}_p$ is the planet's velocity. The magnitude, v_1', is essentially the asymptotic approach speed,

v_{hp}, of Figures 10 and 11. Having specified a launch date and a time of flight for the heliocentric trajectory from Earth to Jupiter, $\bar{v}_1$ is fixed and hence so is the approach velocity, $\bar{v}_1'$. Since the radius of the sphere of influence is also fixed, only two components of the state vector at the spacecraft's entry into Jupiter's sphere of influence remain to be specified. Commonly, this is done by means of an impact parameter $\bar{B}$, which is a vector from the center of the planet normal to the incoming asymptote of the Jupiter-centered approach hyperbola, as shown in Figure 12. The components of the impact parameter in a plane normal to the incoming asymptote $\hat{S}$ then completely specifies the initial conditions. Taking reference axes $\hat{T}$ and $\hat{R}$ in this plane, where $\hat{T}$ lies in the ecliptic plane and $\hat{R} = \hat{S} \times \hat{T}$, the impact parameter is specified in terms of its components $\bar{B} \cdot \hat{T}$ and $\bar{B} \cdot \hat{R}$.

A parametric analysis of the Jupiter-centered hyperbolic trajectories has been carried out for a range of asymptotic approach speeds appropriate to 500-to-600 day Earth-Jupiter trajectories. Values of the semi-major axis $\underline{a}$, the

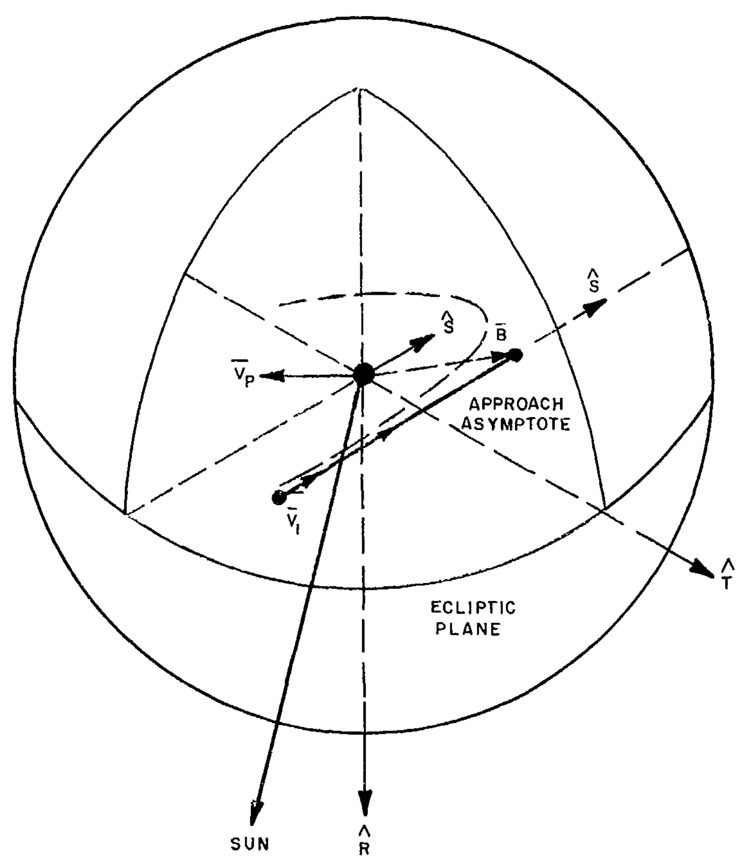


Figure 12. Entry into Jupiter's Sphere of Influence

eccentricity $\underline{e}$, and the asymptote deflection angle γ depend only on the magnitudes of v_{hp} and B. To illustrate the effect of Jupiter's gravitational field the radius of closest approach to the planet, r_p, and the asymptote deflection angle γ, are plotted against B for approach speeds in the range of 10 to 12 km/sec in Figures 13 and 14, respectively. The critical values of $\bar{B}$ which lead to planetary impact lie between 6.6 and 5.2 Jupiter radii (R_j) for the same range of approach speeds.

At exit from Jupiter's sphere of influence, the spacecraft velocity with respect to the Sun is given by

$$\bar{v}_2 = \bar{v}_2' + \bar{v}_p , \tag{3}$$

where $\bar{v}'$ is the spacecraft's velocity with respect to Jupiter at exit from Jupiter's sphere of influence. Inside Jupiter's sphere of influence energy is conserved so that

$$v_1' = v_2' \simeq v_{hp} . \tag{4}$$

If the spacecraft velocity with respect to the Sun at exit from Jupiter's sphere of influence is written as $\bar{v}_2 = v_2 \hat{j}$, then it can be shown that (Ref. 7)

$$v_2 = \bar{v}_p \cdot \hat{j} \pm \left[(\bar{v}_p \cdot \hat{j})^2 + v_{hp}^{\ 2} - v_p^{\ 2} \right]^{1/2} \tag{5}$$

Thus, if the asymptotic approach speed v_{hp} exceeds the planet's speed v_p, the direction of the post-encounter velocity, $\hat{j}$, can be chosen arbitrarily. That is, the post-encounter heliocentric velocity can be pointed in any desired direction — such as towards the Sun or normal to the ecliptic plane. If the approach speed is less than the planet's speed, however, the direction of the post-encounter velocity is restricted by the relationship

$$(\bar{v}_p \cdot \hat{j})^2 \geq v_p^{\ 2} - v_{hp}^{\ 2} \tag{6}$$

or, if the angle between the planet's velocity and the spacecraft's heliocentric velocity is denoted by α, then

$$\sin \alpha < \frac{v_{hp}}{v_p} \tag{7}$$

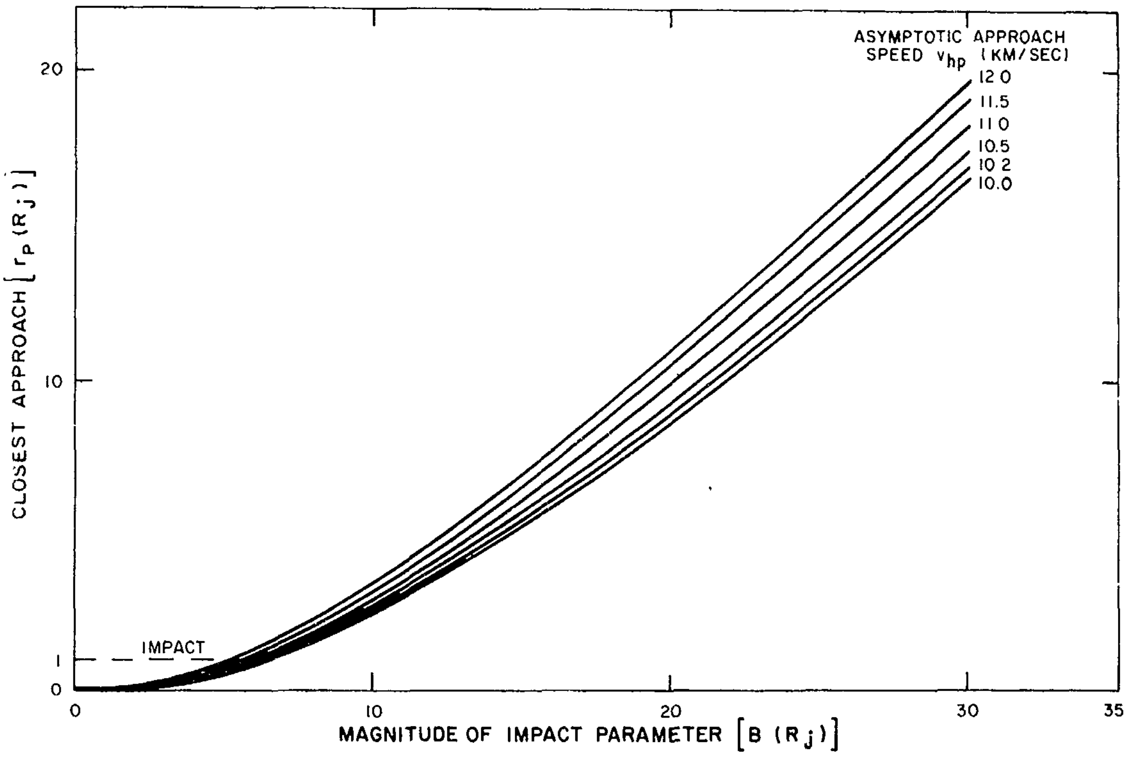


Figure 13. Radius of Closest Approach

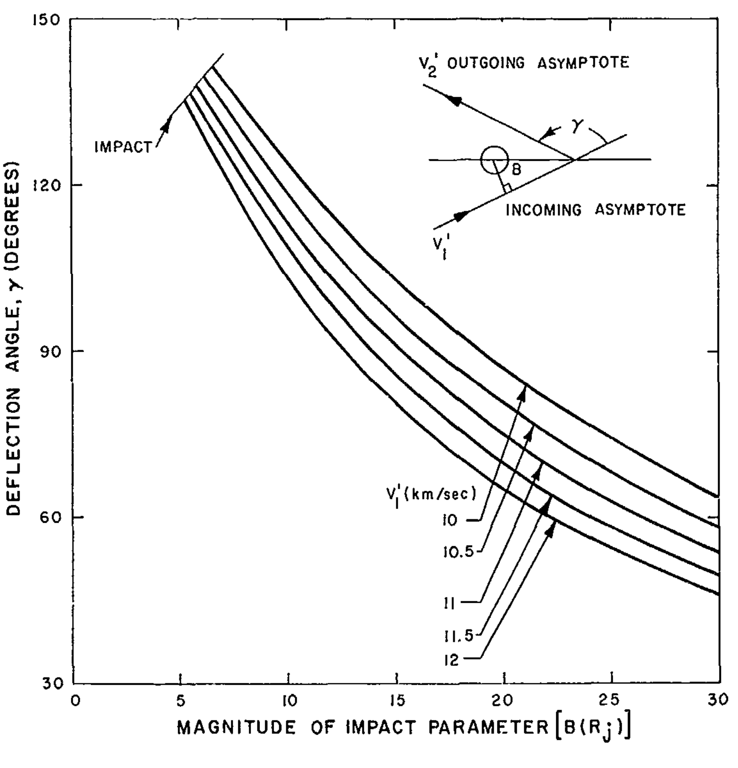


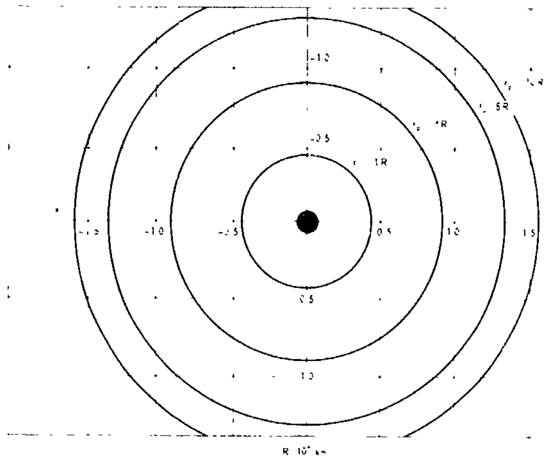
Figure 14. Asymptote Deflection Angle

The spacecraft's velocity is then restricted to a double cone of semi-angle α with the planet's velocity as axis.

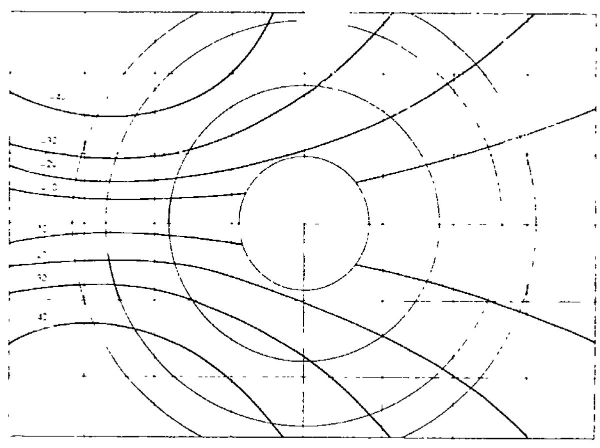
For 550-day trajectories in 1972, the approach speed is 11.14 km/sec and the planet's speed 13.37 km/sec, leading to a maximum post-encounter inclination to the ecliptic of 56.4°. In 1974 the comparable speeds are 10.20 and 13.7 km/sec for a maximum inclination of 48.1°. To achieve an approach speed equal to the planet's speed in order to obtain a post-encounter inclination of 90° to the ecliptic, requires an injection energy $C_3 \simeq 130\ km^2/sec^2$, which is beyond the capabilities of the Atlas/Centaur/TE 364-4 launch vehicle with the baseline GJP spacecraft. Fortunately, the scientific objectives of an out-of-the-ecliptic mission do not require an inclination of 90° but can be met by a trajectory which reaches a distance of approximately 1 AU above the ecliptic plane in the neighborhood of Earth's orbit. By a judicious selection of the aiming point at Jupiter, the post-encounter characteristics can be tailored to meet these requirements.

As the aiming point is moved in the T, R plane the parameters of both the flyby hyperbola and the post-encounter heliocentric orbit are changed correspondingly. Contours of the parameters of interest when plotted on the T-R plane provide a means of selecting an aiming point which simultaneously satisfies several desirable criteria. Such a selection is illustrated in the three views of Figure 15 with numerical values which approximate the 1974, 550-day mission. The scale of the figure is indicated in millions of kilometers on the T and R axes of view A and the solid circle at the origin represents the actual size of Jupiter (R_j = 71,400 km). The strong focussing effect of Jupiter's gravitational field is illustrated by the circular contours of closest approach labelled r_p = 1, 4, 8, 10 R_j. These show that for an aiming point approximately six Jupiter radii from the planet ($B = .42 \times 10^6$ km) the probe will be pulled in to a grazing distance of closest approach. Earth-based measurements of Jupiter's radio emissions in the decimetric range have been interpreted (Ref. 8) as being due to synchrotron radiation in an electron belt with a peak density of 10^7 electrons/ cm^2/sec at 3 R_j. To allow meaningful measurements to be made of the planetary environment without subjecting the spacecraft to possible damaging levels of particle flux, a radius of closest approach between 8 and 10 R_j is appropriate—that is, aiming points should be selected between the outermost circles shown in view A.

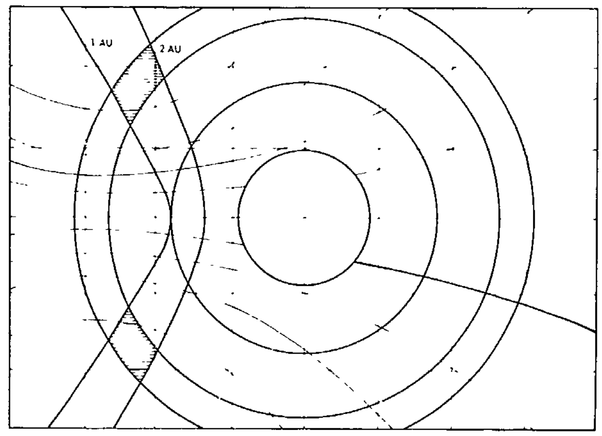
The inclination of the post-encounter trajectory to the ecliptic plane is the second parameter considered in view B of these figures, and contours of i = 10, 20, 30, and 40° are shown. Since the objective of these missions is to reach large distances out of the ecliptic plane, obviously aiming points yielding inclinations of 40°, or more, are appropriate. In conjunction with the requirement for closest approach between 8 and 10 R_j, the post-encounter inclination requirement provides a reasonable delineation of aiming point.



A. Jupiter closest approach contours



B. Post-encounter inclination contours



C. Post-encounter perihelion contours

Figure 15. Aiming Point Selection Criteria (1974)

A further parameter which can be used to localize the most appropriate aiming point is the perihelion distance of the post-encounter trajectory as shown in view C. The baseline spacecraft concept has been developed with a thermal-control system designed to operate in the range from 1 to 10 AU. Preferably then, the out-of-the-ecliptic mission should not approach much closer to the Sun than 1 AU to keep the variation in solar input within tolerable bounds. The 2 AU perihelion contour is not a firm boundary, but it reflects a desire to minimize the time taken to reach the region of maximum scientific interest.

Taken together these three parameters; the distance of closest approach to Jupiter, the post-encounter inclination, and the post-encounter perihelion distance, isolate two small regions of the T-R plane which satisfy all three criterion for an out-of-the-ecliptic mission. These hatched regions of view C are then the prime target areas for out of the ecliptic trajectories. The positive $\hat{R}$ region is preferred since it produces a north-going pass out of the ecliptic plane which improves the spacecraft communication with Northern hemisphere ground stations, such as Rosman, during the period of maximum scientific interest.

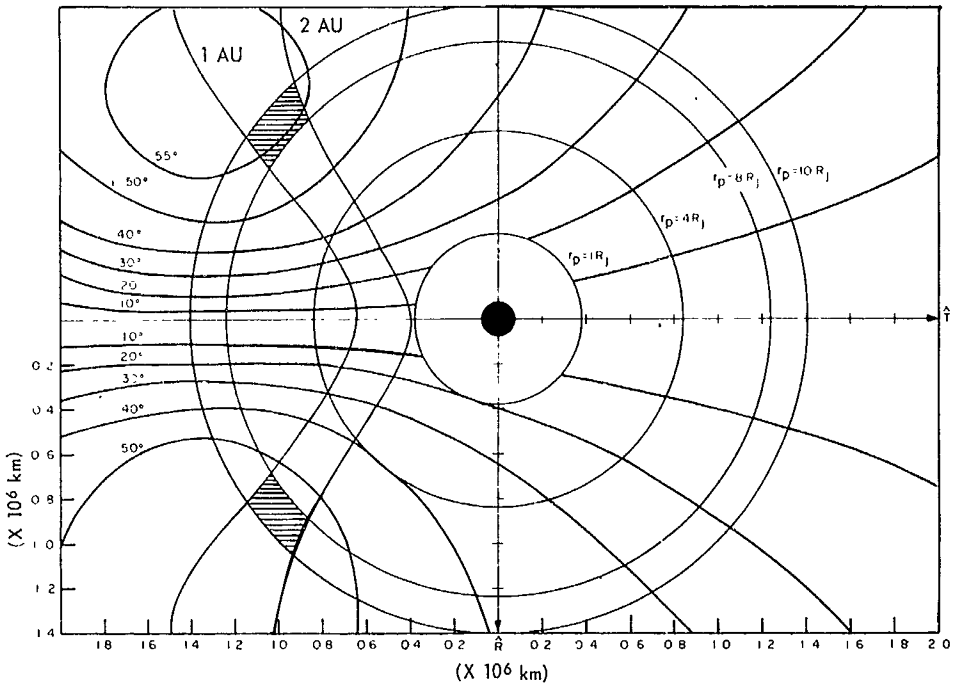
For other missions such as a deep-space probe or a solar probe, different parameters appropriate to the mission objective would be used to select the desired aiming points. For the deep-space probe, for example, contours of the flight time to 10 AU would be an appropriate parameter, and the desire for a minimum flight time provides one criterion for selecting the aiming point.

Composite contour plots for the 1972 and 1974 out of ecliptic missions using 550-day Earth-Jupiter trajectories are shown in greater detail in Figure 16 (A) and (B), respectively.

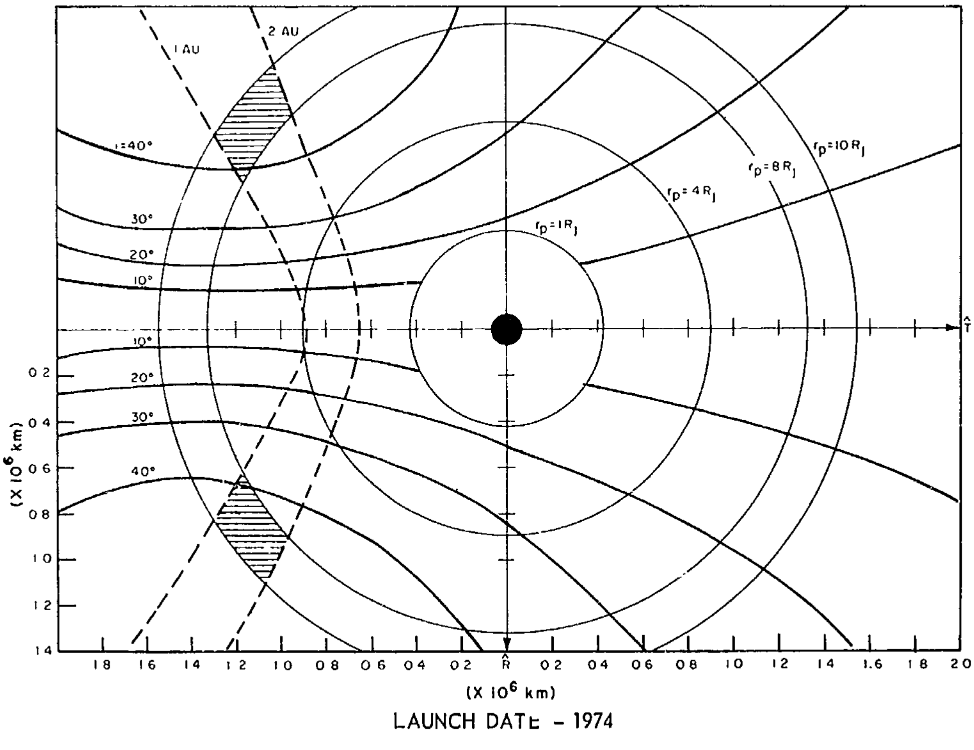
Published figures for launch vehicle injection errors (Ref. 9) lead to 1σ aiming point errors of approximately

$$B \cdot T \simeq \pm 1.25 \times 10^6 \text{ km and } B \cdot R \simeq \pm 0.28 \times 10^6 \text{ km}$$

for 500-to-600 day Earth-Jupiter trajectories (Ref. 1, 6) a 3σ error ellipse would therefore cover the post-encounter contours of Figure 16. A single midcourse correction applied along the spacecraft-Earth line between 10 and 20 days after launch can reduce the in-plane component of miss, $B \cdot T$, to 25,000 km but leaves the out-of-plane component essentially unchanged. The magnitude of the midcourse correction is less than 100 m/sec for the assumed 3σ launch errors and an Earth-line correction in 1972 and 1974. A Sun-line correction is more economical in reducing the in-plane miss ($\sim$80 m/sec) and is less subject to annual variation.



LAUNCH DATE — 1972



LAUNCH DATE – 1974

Figure 16. Post-Encounter Contour for 550-Day Flight

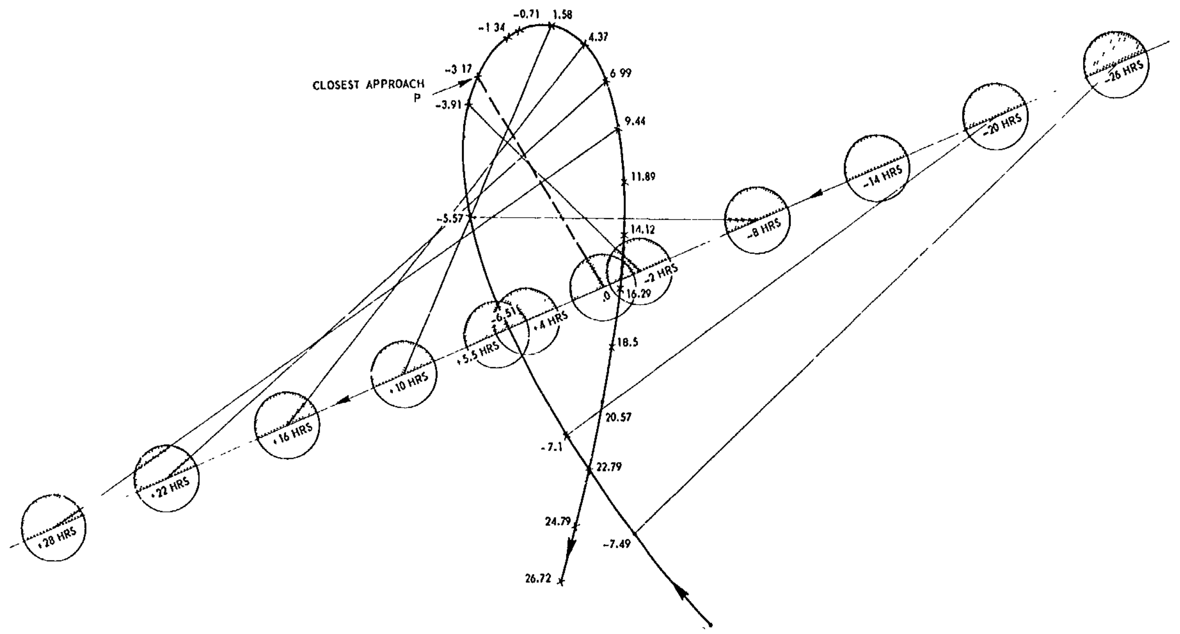
For an in-the-ecliptic mission, the aiming-point requirement to gain a large amount of heliocentric energy and make a quick flight to 10 AU is relatively coarse (Ref. 1), and there is no necessity to correct the out-of-plane launch injection error. For an out-of-the-ecliptic mission, however, the required aiming zones (shaded zones of Figure 16) are sufficiently small to require correction of both in-plane and out-of-plane errors. An arbitrary pointing (optimum) midcourse correction of approximately 100 m/sec can reduce both components of the miss to $\sim$25,000 km. A 3σ error contour would then be approximately the size of Jupiter and would be quite adequate for the present out-of-the-ecliptic mission requirements.

The 25,000-km errors remaining after midcourse correction are due primarily to inaccuracies in predicting thrust level, duration of the burn and direction of the burn. A second midcourse correction applied approximately 100 days after launch with a $\Delta V \simeq 5$ m/sec could reduce the impact parameter errors to $\sim$5000 km. This residual error is due essentially to orbit determination errors, the principal components of which are (1) ephemeris errors, (2) the AU-to-km conversion error, and (3) tracking errors.

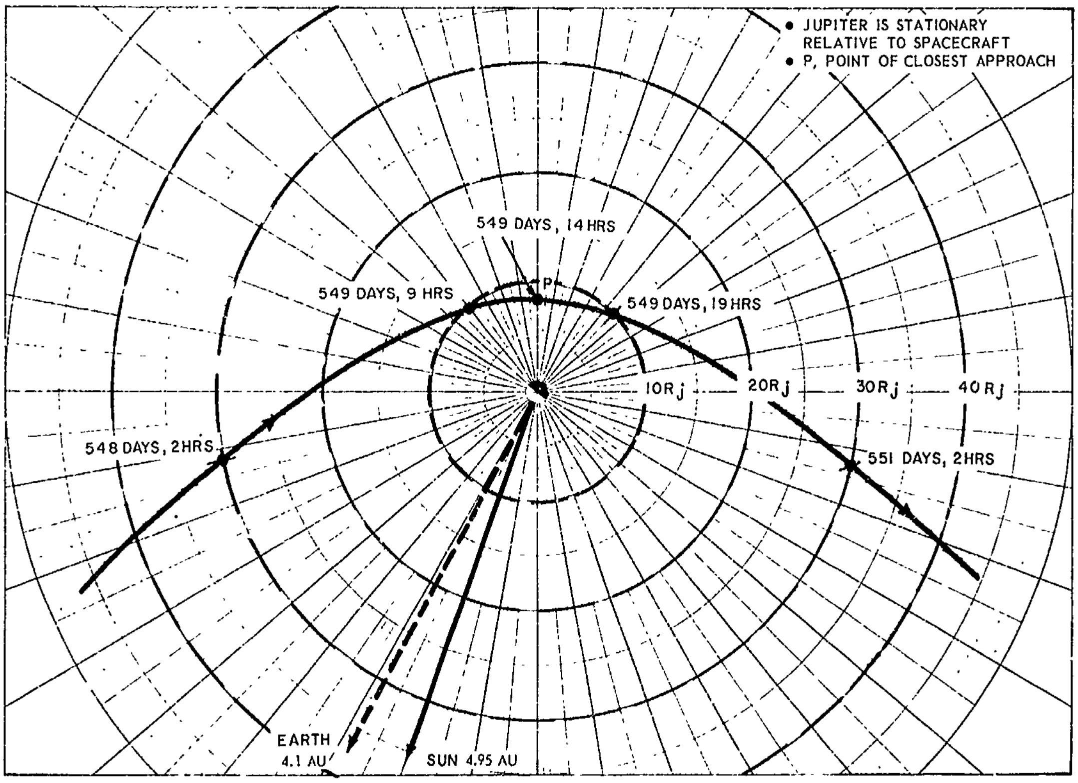
For missions involving a second planet or multiple planets, a second midcourse correction would be advisable as discussed in Section C-2.

Precision n-body trajectories, with nominal flight times of 550 days, show substantial agreement with the post-encounter inclination contours of Figure 16. Such trajectories enter Jupiter's sphere of influence approximately 500 days after launch and spend nearly 100 days there. During this 100 day period of time, the orbital elements of the Jupiter-centered hyperbola change slowly under the influence of the Sun. The changes are not sufficient to be a problem during early mission planning of either the in-the-ecliptic or out-of-the-ecliptic GJP missions. However, they would need to be taken into account for operational trajectory planning and would be quite significant for detailed multiple-planet swingby analysis.

Two views of a typical out-of-the ecliptic trajectory in the neighborhood of Jupiter are provided by Figures 17 (A) and (B). In the first, both planet and spacecraft are shown moving with their proper motion with respect to the Sun. The position of Jupiter is shown at approximately six hourly intervals from one day prior to closest approach to one day after. The corresponding positions of the spacecraft are given by its plan position in the orbital plane of Jupiter together with its altitude above (+) or below (-) the plane indicated in units of Jupiter radii. From this illustration it can be seen that Jupiter threads the eye of the needle formed by the spacecraft's looping trajectory. The position of the Sun is shown by the light and dark hemispheres of the planet which also indicates



A. Projection of spacecraft flyby trajectory onto Jupiter's orbital plane.



B. Jupiter flyby trajectory

Figure 17. Typical Out-of-Ecliptic Trajectory; Jupiter Encounter Phase (1974 out-of-the-ecliptic mission)

the scale of the figure. A more conventional view of the flyby manouevre is obtained by imagining Jupiter to be at rest, that is by viewing the flyby from the position of an observer on the planet. The resulting hyperbolic spacecraft trajectory is shown in Figure 17 (B). Closest approach to the planet on this nominally 550 day transfer occurs 549 days 14 hrs after departure from Earth at a distance of 8.36 Jupiter radii. The spacecraft spends approximately three days within 30 R_j and ten hours within .0 R_j of the planet. Since Jupiter's magnetic field is expected to extend to a distance of approximately 40 R_j towards the Sun an extended period of scientific observation is provided within the region of space dominated by the planet. A third view of the encounter sequence is provided in Figure 18 which shows the projection of the spacecraft's position on the surface of the planet. This shows the latitude coverage of the planet and, since the planet performs a complete rotation in a little less than ten hours, indicates that repe ve longitudinal coverage is available.

The particular trajectory considered here has a post-encounter inclination of 40.4° to the ecliptic plane and the spacecraft reaches a maximum altitude above the plane of ~1.2 AU approximately 1100 days after launch. A time-history of the spacecraft's distance from the ecliptic plane is shown in Figure 19 together with its distance from Earth and the Sun. It can be seen that during the Earth-Jupiter phase of the mission the spacecraft barely leaves the ecliptic (the transfer orbit inclination is ~1.3°), but after encounter it climbs significantly out of the plane reaching 1 AU about 950 days from launch and remaining above 1 AU for n re than 200 days. The spacecraft-Earth distance is close to a minimum at Jupiter encounter, with a communication range of ~4.2 AU. When the spacecraft reaches 1 AU above the ecliptic its communication range is again near a minimum, ~2.6 AU and during this period of maximum scientific interest never exceeds 3.3 AU. Using an 85-foot ground-based receiving antenna, the communication capability of the GJP is approximately 100 bits per second at this distance which should be entirely adequate for the out-of-ecliptic science. The spacecraft-Sun distance is greatest at encounter (4.95 AU) and falls to a minimum of 1.1 AU, shortly before the spacecraft cuts through the ecliptic plane at its descending node. The variation in insolation due to the solar-distance range of 1 to 5 AU is less severe than the 1 to 10 AU range appropriate to the baseline GJP mission. Although the spacecraft remains within 5 AU of the Sun, it spends approximately 800 days beyond 3 AU, at which range RTG power-supply systems are demonstrably lighter than solar-cell systems (Ref. 2). The solar incidence angle is illustrated in Figure 20. Except for the first few days after launch the Sun remains within 25° of the vehicle-Earth axis so that no problem arises from the Sun illuminating the thermal-control louvres.

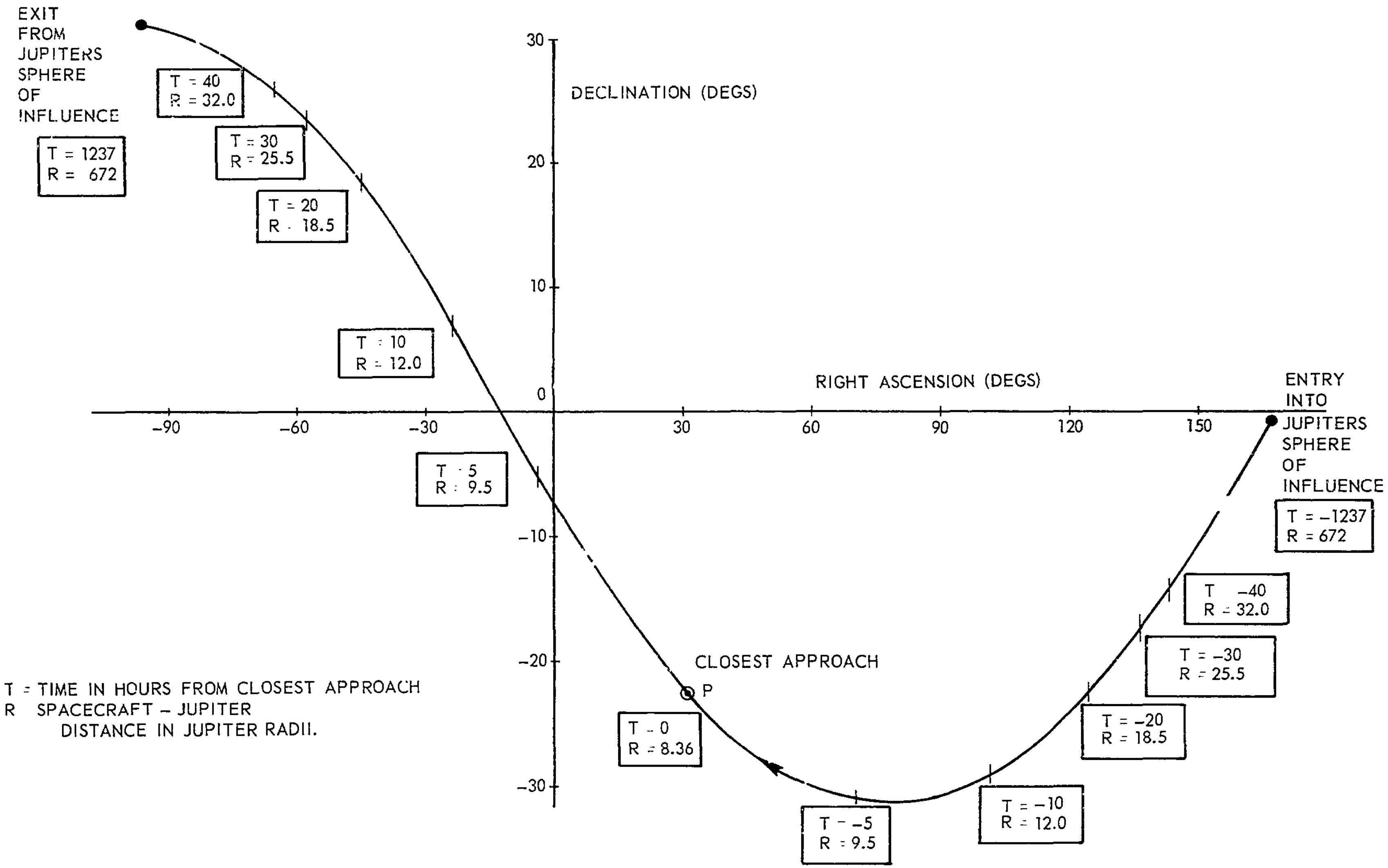


Figure 18. Jupiter Sub-Track During Jupiter Encounter

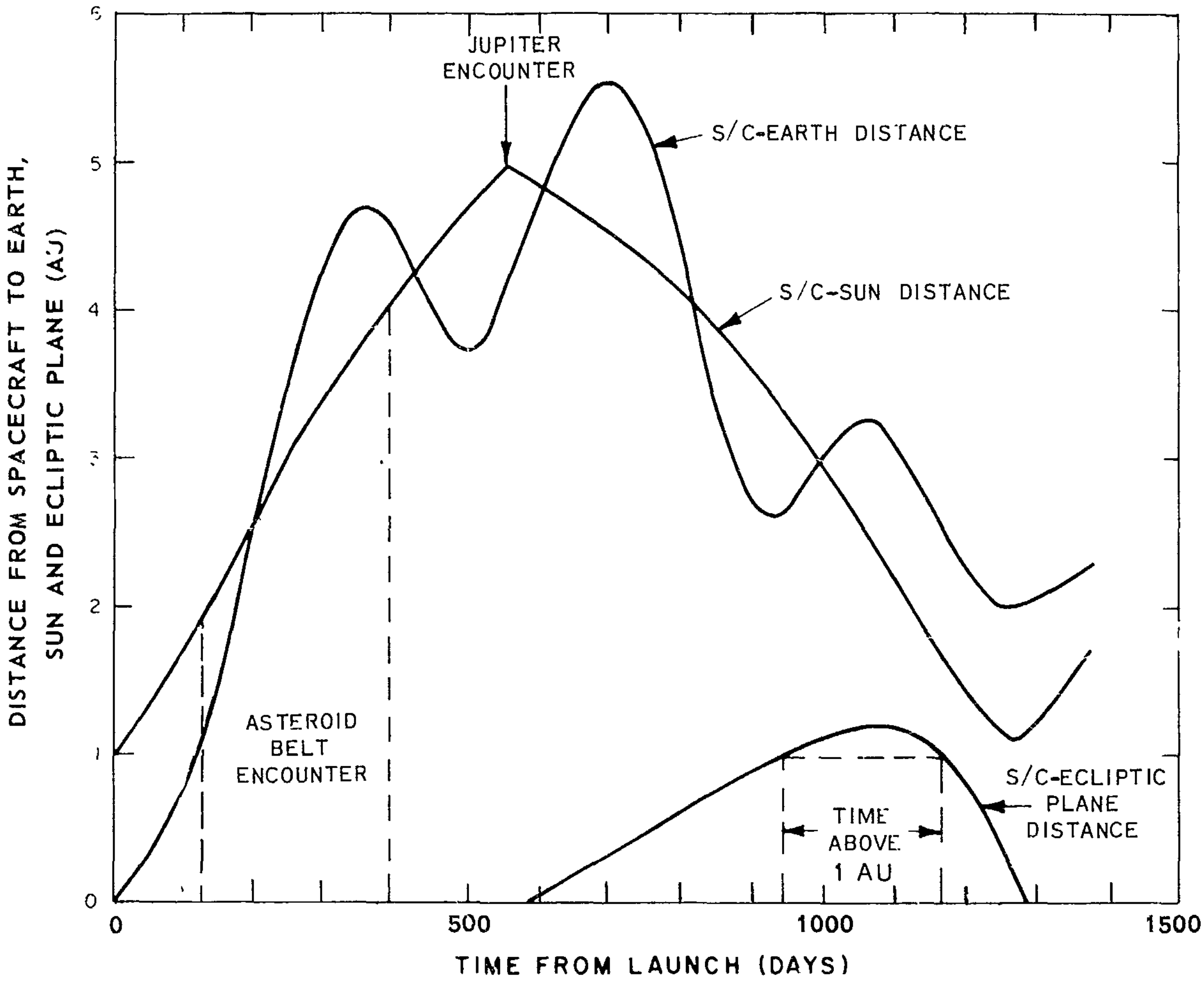


Figure 19. Out-of-Ecliptic Trajectory Distances and Time

C. MULTIPLE PLANET SWINGBYS

Jupiter swingbys to the outer planets Saturn, Uranus, and Neptune, have been investigated (Ref. 10 and 11), and a multiple-planet swingby (of each of these planets in succession), the so-called Grand Tour, (Ref. 12 and 13). These investigations were of a preliminary nature, considering principally the launch opportunities, trip times, and energy requirements and generally avoided the problems of guidance requirements, reliability, communications, and payload. For the most part the referenced studies were concerned with establishing the

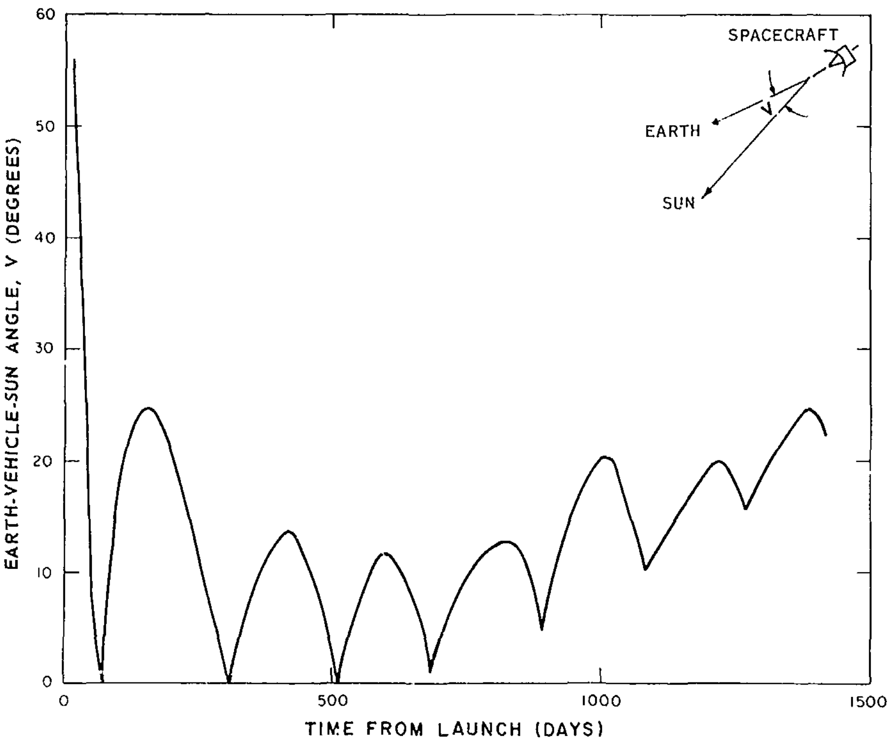


Figure 20. Earth-Vehicle Sun Angle for Out-of-the-Ecliptic Missions

advantage of the Jupiter swingby as opposed to direct missions from Earth to the planets in question. Without exception the referenced study analysis was based on patched conic, or two-body, approximations.

1. The Grand Tour

For the purpose of this study, the Grand Tour is defined as a sequential flyby of the planets Jupiter, Saturn, Uranus, and Neptune at a sufficiently small distance of closest approach for each case to permit scientific observation of all four planetary environments. A typical set of mission objectives is summarized in Table 1. During each of the planetary encounters, attempts would be made to (1) measure the planetary magnetic field and determine its interaction with the solar wind, (2) detect the presence of trapped particle belts and measure their concentrations, (3) measure the composition and physical properties of the planetary atmosphere, and (4) make remote measurements of the planetary surface. The interplanetary measurements to be made during each of the intervening heliocentric legs of the trajectory would be basically similar to those made during the baseline GJP mission.

The closest approach at Jupiter, noted in Table 1, is set by the values of the expected environment, to ensure good measurement capability without endangering the spacecraft. At Saturn, the closest approach should be less than 1.2 planetary radii or greater than 2.3 radii in order to avoid passing through Saturn's rings. Estimates of the particle density in this region (Ref. 12) show that an attempt to fly through the rings would almost certainly be catastrophic. The guidance requirement to pass between the planetary disc and the innermost edge of the ring is extremely severe so that for early missions, at least, passage exterior to the outer ring is preferred. The closest approach at Uranus is set principally by the requirement for a fast flight on the final heliocentric leg of the trajectory rather than any Earth-based prediction of the anticipated environment. Finally a closest approach to Neptune of the order of ten planetary radii is a rather arbitrary but attractive goal.

Annual opportunities for the Grand Tour mission occur during the years 1975 through 1980, due to the favorable geometrical relationship of the outer planets, illustrated in Figure 21. This series of opportunities will not recur for 179 years, being principally set by the synodic period of 174 years between Uranus and Neptune. In Figure 21 the July 1976 launch and the December 1980 launch are used to illustrate the opening and closing of the approaching set of opportunities. The 1976 launch shows the very large deflection of the trajectory that would be required at Jupiter in order to aim for Saturn. In earlier years a suitable deflection is not realizable. The series of opportunities ends in 1980 when the deflection of the trajectory at Jupiter is minimal, indicating a large passage distance and consequently very little gain in heliocentric velocity during the Jupiter swingby. After 1980, Jupiter will move ahead of the outer planets and the Grand Tour will be no longer possible. Figure 25 A and B graphically

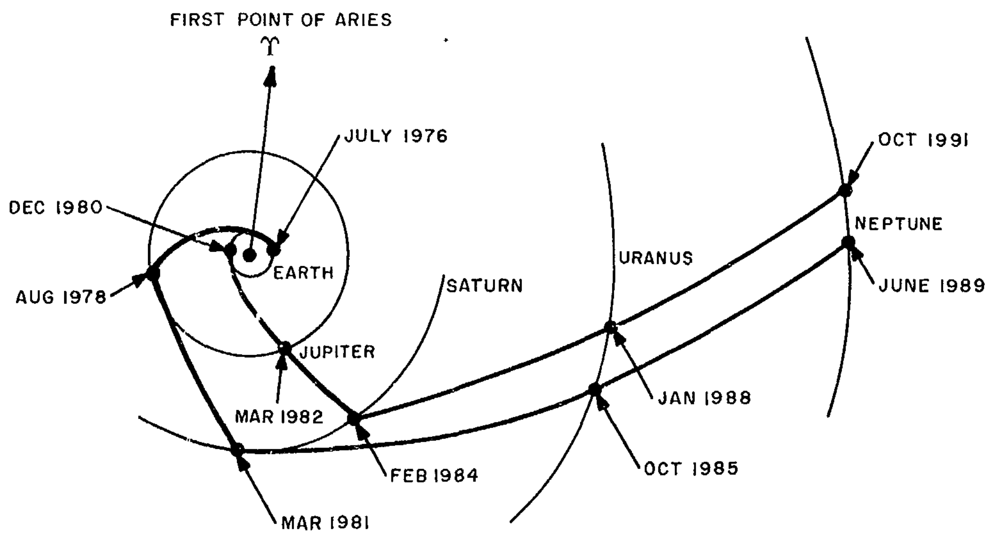


Figure 21. Planetary Geometries for the Opening and Closing of the Grand Tour Opportunities

depict the position of the outer planets with time. While the geometric opportunities exist for the Grand Tour mission during the 1975-1980 period other considerations such as flight time, launch energy requirements and planetary passage distance impose practical limitations on the mission planners in terms of actual launch opportunities.

During the years, 1977 through 1979, satisfactory missions broadly meeting the objectives laid out in Table 1 appear possible. An analysis of such flights has been provided by Silver (Ref. 12); the parameters associated with three example launch dates are summarized in Table 2.

Table 2

Grand Tour Missions

Parameter	Launch Date		
	September 1977	October 1978	November 1979
Launch energy C_3	91.5	102.4	120.9
Jupiter arrival	July 1979	May 1980	March 1981
Closest approach	8.5 R_j	21 R_j	70 R_j
Saturn arrival	July 1981	February 1982	December 1982
Closest approach	2.3 R_s	2.3 R_s	2.3 R_s
Uranus arrival	October 1985	February 1986	December 1986
Closest approach	4 R_u	4.5 R_u	5.5 R_u
Neptune arrival	March 1989	July 1989	July 1990
Total flight time	11.6 years	10.8 years	10.7 years

The September 1977 flight has a launch energy requirement of 91.5 km^2/sec^2, less than the GJP missions described in Section IIA, but with a correspondingly long flight time to Jupiter of 670 days. The passage distance at Jupiter is 8.5 R_j, which is similar to the out-of-the-ecliptic missions previously described, but in this case the trajectory is a trailing-edge flyby of the planet, leading to a large increase in heliocentric energy (so that the probe exceeds solar escape velocity) and a two-year flight from Jupiter to Saturn. A close exterior-ring passage at Saturn yields a four-year flight to Uranus with a closest approach of four planetary radii. The final objective, Neptune, is reached in March 1989 some 11.6 years after leaving Earth.

The October 1978 flight is similar in outline, although the launch-energy requirement has risen to a level slightly above that of the out-of-ecliptic missions. This is indicative of the more distant flyby of Jupiter and the correspondingly lower energy increase which the probe acquires during the swingby. The overall mission duration is reduced to 10.8 years.

For the 1979 mission, the launch-energy requirement is considerably higher (120.9 km^2/sec^2) which indicates the very small effect which Jupiter is exerting on the trajectory, so that essentially all the energy for the Earth-Saturn flight must be provided by the booster. Even assuming no increase in spacecraft weight from the basic GJP spacecraft (which is unlikely from other considerations) the increase in launch energy requirements exceeds the capability of even the Titan IIID/Centaur launch vehicle.

2. Guidance Accuracy Considerations

An estimate of the guidance accuracy requirements at Jupiter for a secondary target planet can be obtained from a simplified analysis of the hyperbolic flyby illustrated in Figure 22. It is easy to show that the departure asymptote angle γ is related to the impact parameter by

$$\frac{\partial \gamma}{\partial B} = -\frac{2}{a e^2} \tag{8}$$

where

a is the semi-major axis and

e is the eccentricity of the hyperbola.

For values of $v_{hp} \simeq 10$ to 11 km/sec, corresponding to 550-day Earth-Jupiter transfers (Figures 10 and 11), and a magnitude of the impact parameter $B \simeq 20\ R_j$ (Figure 14) errors in the deflection angle γ (75° to 85° in this case) and the out-of-plane component due to the clock-angle error $\Delta\Psi$ (Ψ = angle between $\bar{B}$ and $\hat{T}$; near zero in this case) are given by

$$\Delta\gamma = \Delta B/B \tag{9}$$

$$\Delta\Phi = \Delta\Psi \sin\gamma \tag{10}$$

The launch-injection errors lead to (1σ) errors in the impact parameter at Jupiter of $\Delta B \cdot T = 1{,}250{,}000$ km and $\Delta B \cdot R = 228{,}000$ km, as indicated in Line 1 of Table 3. These errors were seen to be intolerable even for the baseline GJP deep-space mission or the out-of-ecliptic mission described in Section IIB. If the spacecraft is tracked from Earth during its first few days of flight and its actual orbit computed, a single midcourse correction of magnitude $\Delta V \leq 100$ m/sec can be applied along the spacecraft-Earth line which will reduce the in-plane component to 25,000 km while leaving the out-of-plane component essentially unchanged. This approach, shown in Line 2 of Table 3, is sufficient for the baseline GJP mission but is not adequate to perform an out-of-the-ecliptic mission. If the spin axis of the spacecraft is moved from its Earth-pointing orientation to the optimum direction in space, then a $\Delta V \leq 100$ m/sec can reduce both components of the miss to the order of 25,000 km (Line 3, Table 3). This accuracy was previously seen to be appropriate to the out-of-the-ecliptic mission guidance requirements. Applying these same errors (Line 3, Table 3) to

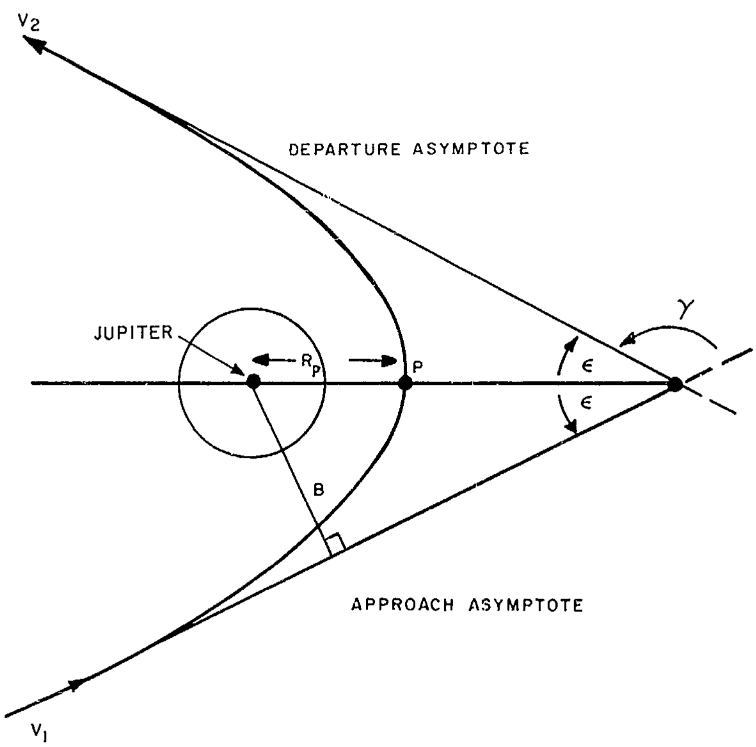


Figure 22. Jupiter - Centered Hyperbola

Table 3

Summary of Trajectory-Correction Requirements

Applicable Mission	Pre-Jupiter Encounter Correction			Jupiter Errors			Post-Jupiter Encounter Correction	Saturn Errors
	10 Days (m/sec)		100 Days (m/sec)	B · T (km)	B · R (km)	$\Delta\theta$ * (degrees)	(m/sec)	(R_s)
	Earth Line	Arbi-trary	Arbi-trary					
1. (Launch)	$0^{\dagger}$	-	0	1.25×10^6	$.288 \times 10^6$	-		
2. Baseline GJP (Deep-Space In-ecliptic)	100	-	0	25,000	$.288 \times 10^6$	-		
3. Out-of-Ecliptic	-	100	0	25,000	25,000	1°		
4. GJP with Arbitrary Pointing 1 pre & 1 post en-counter correction	-	100	0	25,000	25,000	1°	230	2
5. GJP with Arbitrary Pointing 2 pre encounter corrections	-	100	5	5,000	5,000	0.2°	0	40
6. Two-Planet Swingby	-	100	5	5,000	5,000	0.2°	46	0.5

*Error in direction of departure velocity at Jupiter.

† ATLAS/CENTAUR/TE 364-4 Injection errors.

the multiple-planet swingby mission, it can be seen that for a nominal value of $B \simeq 1.5 \times 10^6$ km the error in the departure asymptote angles γ and Φ are nearly 1°. If these pointing errors were allowed to propagate along the Jupiter-to-Saturn heliocentric phase of the swingby, they would lead to impact-parameter errors of approximately 1.25×10^7 km, i.e., 200 Saturn radii (R_s). Saturn's sphere of influence has a radius of approximately 5.5×10^7 km, therefore, such a trajectory would be expected to pierce it, but the errors are clearly intolerable for all but the most elementary mission objectives. If errors in the Jupiter departure velocity of this magnitude were determined from Earth-based tracking and a post-encounter midcourse correction applied to null them, its magnitude would be $\Delta V \leq 230$ m/sec (Line 4, Table 3). Inaccuracies in applying the desired ΔV would lead to impact-parameter errors of the order of 2 R_s, which would dominate the orbit determination residuals of approximately 10,000 km.

If a second pre-Jupiter-encounter correction were applied about 100 days after launch ($\Delta V \sim 5$ m/sec, Line 5 of Table 3), the uncertainty in the Jupiter impact parameter could be reduced to essentially the orbit-determination errors of 5000 km. The corresponding Jupiter-departure velocity errors would be proportionately reduced ($\Delta\gamma = \Delta\Phi = 0.2°$) and, if allowed to propagate along the Jupiter-Saturn path, would give rise to Saturn impact parameter errors of ~ 40 R_s. This could provide a relatively simple two-planet swingby mission with no requirement for the spacecraft to assume an arbitrary orientation while at a large distance from Earth to perform a post-Jupiter-encounter trajectory correction, and with a total ΔV capability very similar to GJP. If the post-encounter trajectory is determined from Earth-based observation and a correction applied for velocity errors of the calculated magnitude, then ΔV would be approximately 46 m/sec. The residual errors due to inaccuracy in the midcourse correction and tracking uncertainty would be approximately 0.5 R_s (Line 6, Table 3). This is sufficiently accurate to allow meaningful scientific observations to be made of the planet, but would not allow for an interior-ring passage at Saturn. The 0.5 R_s accuracy at Saturn encounter is much too coarse to provide a controlled close flyby of either Uranus or Neptune. Additional mid-course corrections or, preferably, planetary-approach guidance, would be required to perform the Grand Tour mission defined here. A useful, though less ambitious, two-planet swingby appears to be more appropriate extension of the baseline GJP mission capabilities.

3. Two-Planet Swingby

The Grand Tour mission described in Section C-1 provides an opportunity to visit all four of the outer planets using a single spacecraft and launch vehicle. During the 1977 and 1978 launch opportunities the launch energy required is relatively small, comparable to GJP mission, due to the large gravity assist provided by Jupiter. Nominal trajectories for these opportunities yield sufficiently close flybys of each planet to permit meaningful scientific investigation of its environment. The guidance requirements for these missions however are extremely demanding and will almost certainly require the development of a sophisticated on-board planetary approach guidance system.* In addition the opportunities for these missions are very restricted as they depend on the favorable location of all four outer planets. A pattern similar to that which makes the Grand Tour possible in the late 1970's does not recur until the year 2154.

Separate direct flights to each of the planets provides an alternative means for outer planetary exploration but requires a separate spacecraft with its associated launch vehicle for each mission. The guidance requirements for each mission could be satisfied by an Earth based tracking system similar to that proposed for the basic GJP. Annual opportunities exist for the direct flights, however the launch energy requirements for Saturn, Uranus and Neptune missions are all considerably in excess of the GJP Jupiter flyby.

A third method of probing the environment of Saturn, Uranus, Neptune and possibly Pluto† also, is to perform a series of two planet swingbys. Such missions provide most of the benefits of the Grand Tour while not requiring the development of on board guidance systems. By making use of Jupiter's gravity assistance they require less launch energy than direct flights and the opportunities for launch occur much more frequently than for the Grand Tour. In addition a series of three or four such flights, spread over a reasonable time scale, would provide all the planetary data and a more comprehensive temporal sampling of interplanetary space than would be provided by a single Grand Tour mission. Simultaneous measurements made from two or more probes at different locations would enable temporal and spatial variations in the interplanetary measurements to be differentiated.

Of the outer planet missions considered here, the two-planet swingby via Jupiter, using an Earth-based tracking system for guidance with two pre-Jupiter-encounter and one post-Jupiter-encounter trajectory corrections, appears to be the most promising. The launch-energy requirement of the 1977 Earth-Jupiter-Saturn mission is within the capabilities of the proposed SLV3C/Centaur/TE 364-4 booster, and the mission requires relatively minor modifications to the basic

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*Recent studies performed by JPL (Scull, J. R. AIAA Paper No. 68-1105, Oct. 1968) show a requirement for 9 trajectory corrections with a total ΔV of 317 m/sec (147 lb. of propellant) for a Grand Tour mission utilizing onboard guidance compared with 11 trajectory corrections with a total of 1857 m/sec (1000 lb of propellant) for an Earth based system.

†In the general time period of interest Pluto will be within Neptune's orbit. See Figure 25 for in-plane location. Trajectory corrections can be made near Saturn, for example, to deflect the spacecraft's trajectory sufficiently out-of-the-ecliptic plane to intersect Pluto's orbit.

GJP trajectory correction capability. Although it has not been possible to perform a guidance analysis of other two-planet swingbys in this study it appears that a similar approach would be feasible and would provide an attractive GJP-growth capability worthy of further study. Use of a more energetic and more accurate launch vehicle, such as the SLV3X/Centaur/Burner II or the Titan IIID/Centaur, would extend the capabilities to a range of two-planet swingbys during the late 1970's and early 1980's with trajectory-correction requirements within the basic GJP capabilities.

While opportunities for direct trajectories to the outer planets occur yearly, the opportunities for missions using a Jupiter swingby occur only at the frequency of the synodic period between Jupiter and the target planet, although these opportunities last for 3 to 5 years. Opportunities for Jupiter swingby missions to each of the outer planets occur in the 1975 to 1980 period. The opportunity then ends because Jupiter moves ahead of the outer planets. Subsequent opportunities for these missions occur in regular cycles approximately as shown

Jupiter - Saturn	1976 - 1980 and then 1996 - 2000 and so on
Jupiter - Uranus	1978 - 1982 and then 1992 - 1996 and so on
Jupiter - Neptune	1978 - 1982 and then 1991 - 1995 and so on

It is still possible to realize Saturn swingbys to Uranus and Neptune during the 1980's.

Typical results comparing swingby and direct flights to Saturn, Uranus, and Neptune are shown in Figures 23 A and B and 24.* In summary, Jupiter-swingby trajectories to any of the outer planets can be accomplished for a launch energy close to the minimum Earth-Jupiter requirement, $C_3 = 82\ km^2/sec^2$, (Figures 2 and 3) although the corresponding flight times are very long. For C_3 near 100, giving Earth-Jupiter flight times $\sim$550 days, swingby trajectories to Saturn require total flight times of approximately 3 years, to Uranus about 6 years, and to Neptune about 9 years. Direct flights to the outer planets require C_3's greater than 100, and for energies where direct flights first become possible, swingby trajectories offer considerable flight-time savings.

Figure 25 shows the orbit of the planets Jupiter, Saturn, Uranus, Neptune and Pluto for the years 1972 through 1999 and identifies the planet position as of January 1 of the year noted. For the angular coordinate the reference zero point, by convention, is taken as the first point of Aries (♈).

*Based on References 10, 11, 12 and 13.

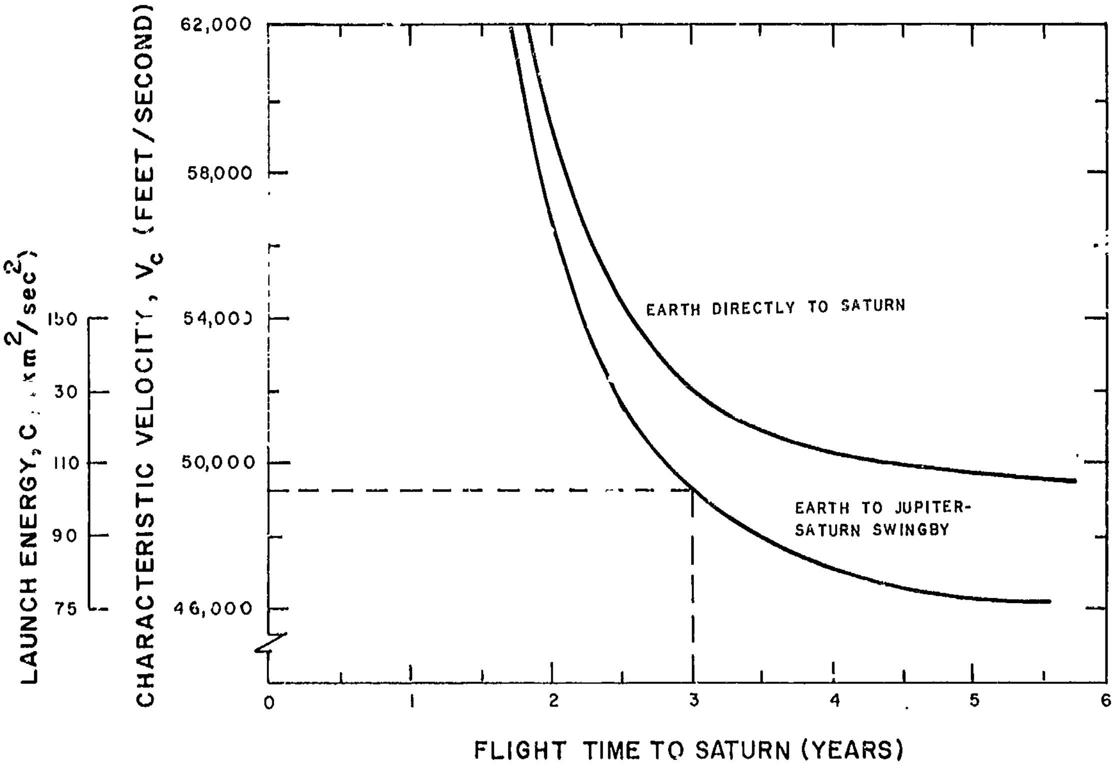


Figure 23A. Comparison of Two-Planet Swingby and Direct Flight to Saturn.

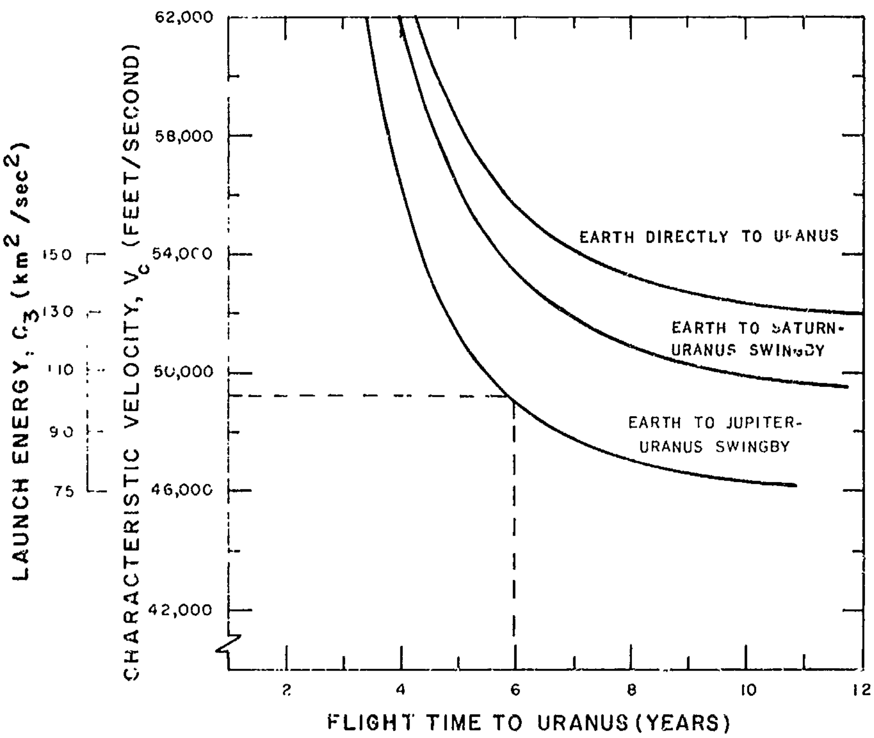


Figure 23B. Comparison of Two-Planet Swingby and Direct Flight to Uranus

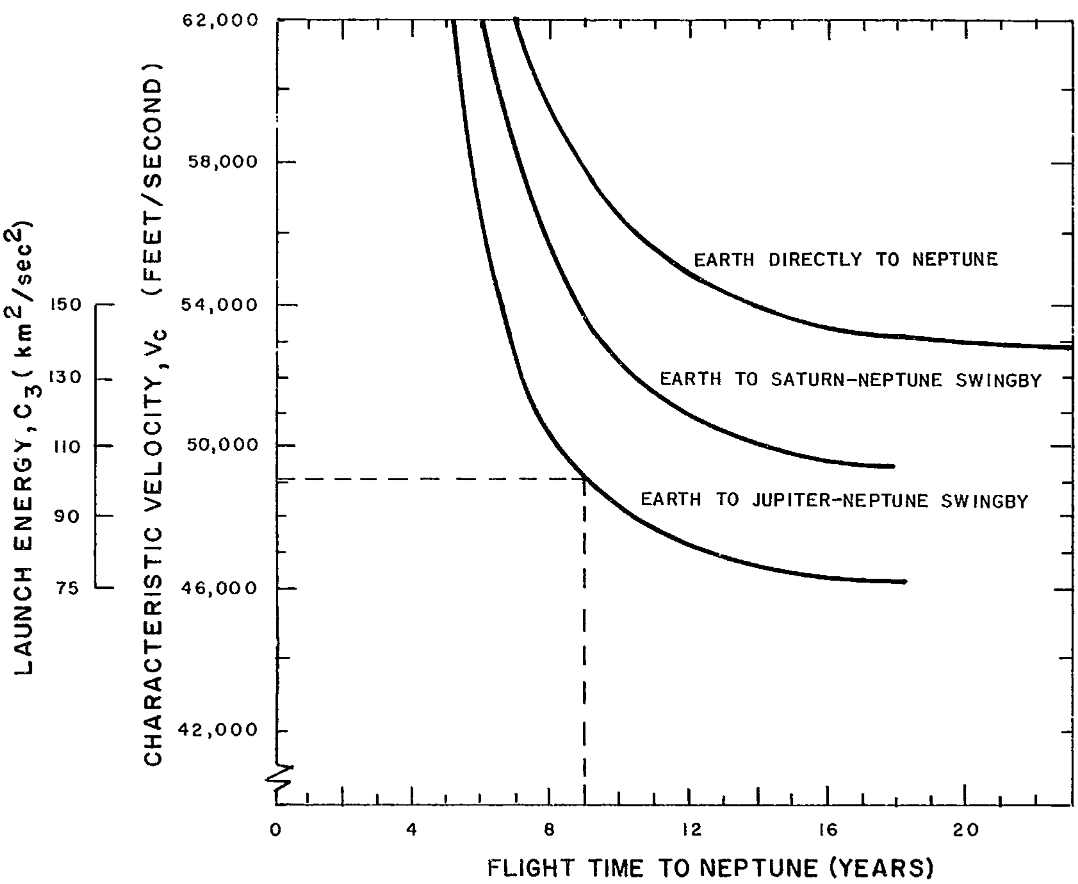
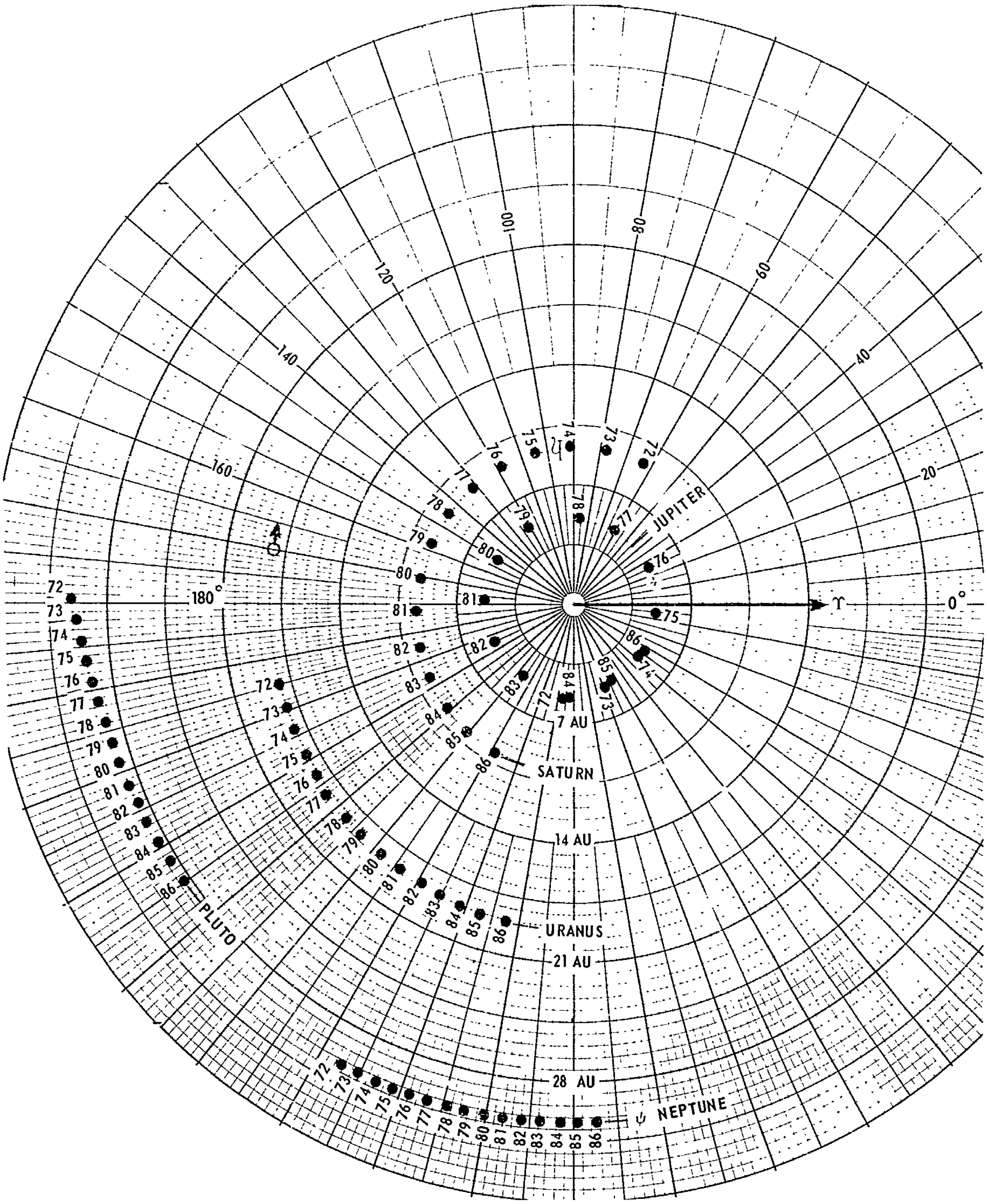
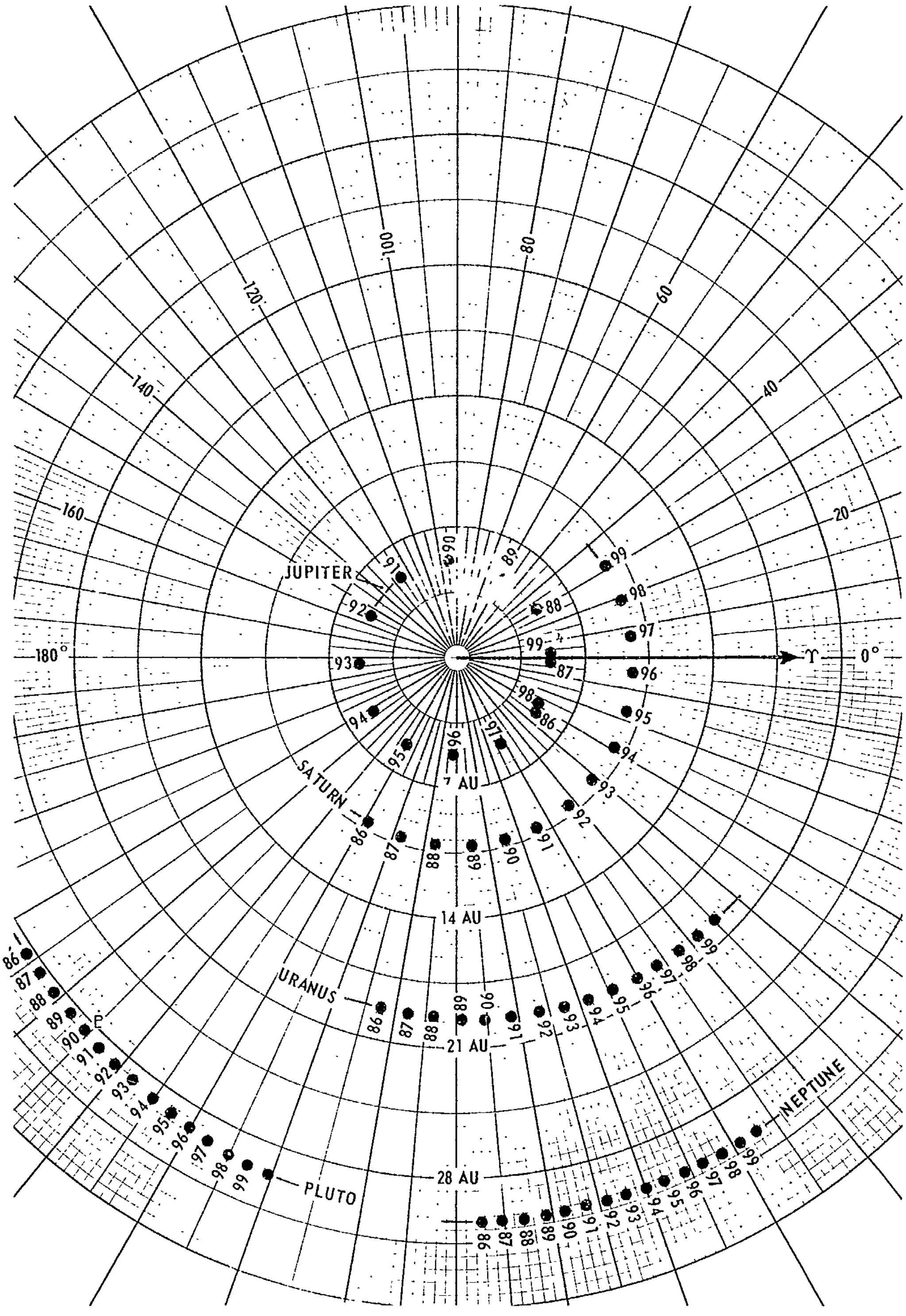


Figure 24. Comparison of Two-Planet Swingby and Direct Flight to Neptune



View A. Years 1972 Through 1986

Figure 25. Orbits of the Outer Planets



View B. Years 1986 Through 1999

Figure 25. Orbits of the Outer Planets (Continued)

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SECTION III

LAUNCH VEHICLE CAPABILITIES

A. SELECTION OF LAUNCH VEHICLES*

During the course of this study it became apparent that the potential list of launch vehicles is, in practice, restricted. Previous studies have considered possibilities ranging all the way from the Saturn V/Centaur to the Atlas SLV3C/Centaur, including the Saturn I and Titan III variants between. Since relatively simple precursor missions have the most interest in the present context, many of these launch vehicles are impracticable.

The rather modest aims inevitably associated with precursor flights are not such as to demand, or justify, very large payloads, therefore, a limited booster capability is acceptable. The range of possibilities receiving detailed examination is thereby drastically reduced to include the Atlas SLV3/Centaur/Kick family and the Titan IIID/Centaur combination. The kick stage for the Atlas family is, in practice, inevitably a solid-fuel engine of the TE 364-series.

The Titan IIID/Centaur vehicle has been proposed as offering not only a large, immediate, high-velocity capability, but also growth possibilities due to the development work proceeding on the basic Titan vehicle. Integration of the Titan and Centaur vehicles is expected to be completed in time for the Mars '73 mission so that its consideration for the Grand Tour or two-plant swingby in the late 1970's is appropriate.

For the first two model missions, attention has been focused upon the Atlas variants. The standard Atlas/Centaur vehicle, as employed for current missions, includes the SLV3C lower stage; the booster elements are characterized by

SLV3C

Booster thrust	336,000 lb	Usable propellants	268,000 lb
Sustainer thrust	58,000 lb	Gross launch weight	287,000 lb

*See Appendix III, which is a reprint of Advanced Plans Staff Paper 69-2 "Launch Vehicle Considerations For Developing An Outer Planets Exploration Strategy" by George M. Levin dated Feb. 1969 covering other launch vehicles.

Centaur

Thrust	30,000 lb
Usable propellants	30,500 lb
Gross launch weight	37,600 lb

It should be noted that the value given for Gross Launch Weight of the Centaur Stage includes an arbitrary nose fairing, insulation panels, and some allowance for boil off. In addition, the Centaur propellant load includes about 500 pounds of nonpropulsive but expended material, so that "usable" should not be confused with "useful."

Uprated versions of both booster elements have been proposed for other missions. Of these, the most significant change is represented by the Atlas SLV3X development; however, this is not presently an available Launch Vehicle for NASA missions but, as proposed, it involves replacement of the MA5 engines in the booster stage with higher performance H1 engines and an increase in the propellant tank length. While this development is feasible, the program would require a cost effectiveness analysis. The Centaur Stage is evolving as time goes on with improved insulation as a major change; this will allow coast times between Centaur burns to be extended initially from the present 25-minute limitation up to one hour and eventually to as much as six hours (444 and 452 Series). Additionally, the elimination of the boost pump and provision of throttleable engines are also proposed modifications. The definitive version of the uprated Centaur is currently termed Centaur 70.

For the necessarily high geocentric launch energies associated with deep-space missions, the 2 1/2 stage Atlas/Centaur stack is inadequate alone and the use of a kick stage is mandatory. Given the requirement for such a stage, the possibility exists for the development of a special-purpose high-efficiency unit, as has often been proposed in the past for high-velocity missions. However, it emerges that useful, though not maximal, performance in the velocity range up to 50,000 ft/sec can be achieved by the employment of a conventional solid-propellant motor, of the TE 364 Thiokol series. The characteristics of the -3 and -4 versions of this engine are listed below.

TE 364-3

Gross motor weight	1,578 lb
Weight at "all burnt"	124 lb
Total Impulse	417,500 lb/sec

TE 364-4

Gross motor weight	2,244 lb
Weight at "all burnt"	129 lb
Total Impulse	602,400 lb/sec

The combined predicted performance of the various Atlas/Centaur/TE 364 combinations is shown in Figure 26. This reflects the latest (Ref. 1) information available on the SLV3X and also on the Titan IIID/Centaur.

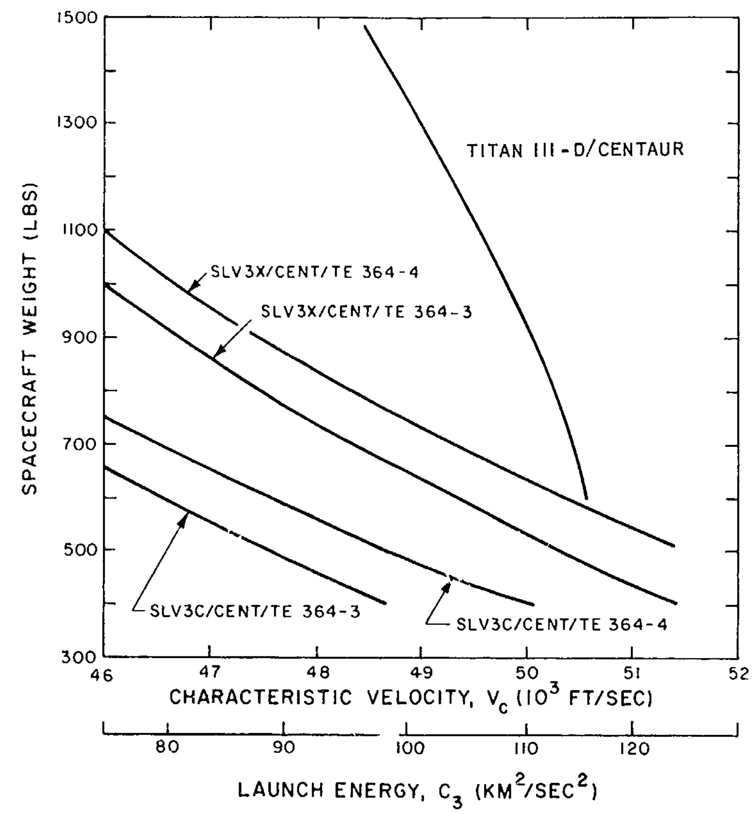

Figure 26. Launch Vehicle Capabilities

It is apparent from the data given in Figure 26 that, for precursor deep-space or out-of-the-ecliptic missions, the SLV3C/Centaur/TE 364-4 combination offers respectable performance for the cost of the development of the -4 Thiokol engine from the existing -3 model. The performance advantage to be gained from use of the -4 rather than the -3 solid kick stage is significant and worthwhile, whichever Atlas variant is considered.

So far as growth missions are concerned, it would ultimately be necessary to move to a larger vehicle since the capability of the SLV3C/Centaur/TE 364-4 is not such as to allow substantial payload uprating beyond the minimal precursor requirements. If the SLV3X stage becomes available, its use would offer an increased payload capability of about 250 pounds for 550-day trajectories. Alternatively, if the spacecraft weight can be held to approximately 600 pounds, the SLV3X/Centaur/TE 364-4 provides a characteristic velocity capability in excess of 50,000 ft/sec. This increase in characteristic velocity fits the requirements of the Jupiter-Saturn, Jupiter-Uranus and Jupiter-Neptune "Mini-tours," discussed in Section II-C. Use of the SLV3X first stage, though not necessary for the precursor missions, provides considerable flexibility for growth with respect to a variety of two-planet swingbys. The Titan IIID/Centaur is capable of supporting payloads in excess of 1200 pounds to a characteristic velocity of 49,000 ft/sec. This makes it an alternative for the two-planet swingbys and the most likely candidate for a Grand-Tour Mission in 1977 or 1978, which appears to require a much larger spacecraft than the baseline GJP. For very high velocity missions ($v_c \geq$ 50,500 ft/sec), however, its payload capability falls below that of the SLV3C/Centaur/TE 364-4.

As a result of an appraisal of the candidate launch vehicles, and of the payloads associated with the precursor missions of interest, it is possible to "select" boosters for the model missions:

(1)	1972 GJP Mission out-of-ecliptic:	SLV3C/Centaur/TE 364-4
(2)	1974 GJP Mission out-of-ecliptic:	SLV3C/Centaur/TE 364-4
(3)	Two-Planet Swingby:	SLV3X/Centaur/TE 364-4 or Titan IIID/Centaur
(4)	Grand Tour (1977, 1978):	Titan IIID/Centaur

Later work performed at GSFC suggests the use of the SLV3X/Centaur with a Burner-II upper stage. The use of this guided upper stage reduces the on-board trajectory correction requirements for a two-planet Jupiter swingby to a level within the baseline GJP capabilities.* The payload capability of the vehicle, however, is necessarily less than the SLV3X/Centaur/TE 364-4 so that the choice of launch vehicle for these growth missions involves some compromise.

*As this report was being edited, the SLV3X is not in NASA's launch vehicle development program. However, the Titan IIID/Centaur, which has been identified as an alternative launch vehicle for the two-planet swingbys, also provides the improved injection accuracy associated with a guided upper stage and has ample payload capability.

B. SLV3C/CENTAUR/TE 364-4

The assumed vehicle/upper-stage characteristics, in terms of weight (in pounds) will be

Centaur, dry	4,005
Useful propellant	29,911
Non-propulsive Centaur expendables	491
Centaur/TE 364 adapter	162
Delta spin table	175
TE 364-4, loaded	2,244
Delta adapter	20
Spacecraft	600

where the upper limit of the range of spacecraft weight has been taken to allow for a reasonable margin in performance.

These assumptions are fairly firm, as are the specific impulse values which may be set at 440 seconds for the Centaur and 287 seconds for the solid stage. The principal difficulty, in the absence of specific data from the booster manufacturer, is in the identification of the Centaur separation velocity. By back tracking from existing data, and for the purpose of the calculation, this velocity has been set at 12,400 ft/sec, and the mission profile, shown in Table 4, emerges, allowing for chill down, boil off, and so on.

Table 4

SLV3C/Centaur/TE 364-4 Mission Profile

Mission Stage	Weight (lbs)	Velocity (ft/sec)
Centaur first ignition	37,550	12,400
Centaur first shut off	16,310	24,190
Centaur second ignition	15,950	24,225
Centaur burn out	7,210	35,485
TE 364-4 ignition	2,864	35,485
TE 364-4 burn out	749	47,785
Spacecraft separation	600	47,785

The Centaur burn-out velocity of 35,485 ft/sec compares well with the value of 35,400 ft/sec, obtained in the Phase A Report (Ref. 1), and the final velocity of 47,785 agrees with the value given in Figure 26. The above data are, therefore, substantially correct and has been used for preliminary analysis of abort modes in the Task VII-B Report, "Spacecraft/RTG Feasibility Study."

SECTION IV

SCIENTIFIC OBJECTIVES

A. GENERAL

The planetary and interplanetary environment has been well specified by Dr. J. Trainor in the GJP Phase A Study (Ref. 1) and although refinements are likely to be made from time to time as a result of near-Earth measurements, many of the uncertainties in the present models will not be resolved until suitable direct or near flyby measurements are performed by a GJP mission.

The general areas of scientific interest in a Jupiter gravity-assist mission are

(1) Extending knowledge of the interplanetary medium to Jupiter-distance from the Sun and either continuing in the plane of the ecliptic to 10AU and beyond, or providing a latitude profile of the environment in an out-of-the-ecliptic plane.

(2) Obtaining a close view of the planetary environment of Jupiter during the encounter, and other planets as appropriate.

(3) Improving knowledge of astronomical constants such as the AU-to-km conversion and the accuracy of planetary ephemerides.

B. ENVIRONMENTAL PHENOMENA DURING INTERPLANETARY PHASE

During the interplanetary phase of the mission, either in or out of the ecliptic plane, the environmental phenomena of interest are

(1) The spatial and temporal variation of the solar wind and the associated magnetic field.

(2) Galactic cosmic rays.

(3) Solar flares and cosmic rays.

(4) Meteoroid flux.

1. Solar Wind

At the present time the solar wind and its magnetic field are being monitored at the orbit of Earth and a history of such data is being built up over a solar cycle. The solar wind consists of a neutral gas composed principally of highly ionized hydrogen (~90%) and helium (~10%) with plasma characteristics—density 1 to 10 particles/cm^2, streaming velocity 300 to 600 km/sec, thermal speed 30 to 60 km/sec, magnetic field $\sim 5\gamma$. In addition to the near-Earth environment, measurements have been made between the orbits of Earth and Venus, and between Earth and Mars during the flights of Mariner vehicles. Extrapolation of these data to a large solar distance, or to a significant distance from the ecliptic plane is unreliable. A primary objective of the mission is to extend the range of observation, perhaps to the limit of the organized solar wind, or at least so that the galactic boundary can be reasonably predicted.

2. Galactic Cosmic Rays

Galactic cosmic rays consist of atomic nuclei whose relative abundance roughly parallels the estimated cosmic abundance of these atoms. Measurements made at 2 BeV per nucleon show 94% protons, 5.5% alpha particles, and a remainder of heavier nuclei up to atomic number 28. Their energies (up to 10^{19} eV) extend beyond the range which could be produced by solar processes (~10^{10} eV). The intensity of galactic cosmic rays at the orbit of Earth is modulated by solar activity, rising to a maximum at times of solar minimum. Measurements of the gradient of cosmic-ray intensity as a function of rigidity over a limited heliocentric range (~1.5 AU) indicates that the Sun's influence extends to a range of 10 to 100 AU. Similar measurements on a Jupiter swingby mission should give a much more precise estimate of the boundary and of the interstellar intensity of cosmic rays.

3. Solar Flares and Cosmic Rays

Solar cosmic rays have their origin in solar flares. They consist primarily of protons and alpha particles and occasionally higher Z components. Their energies are at the lower end of the cosmic-ray energy spectrum with about five relativistic (> 1 BeV) events occurring, on an average, during an eleven-year solar cycle. The Sun emits sufficient non-relativistic particles to interfere with Earth surface communications about ten times a year.

4. Meteoroid Flux

Approximately 90 percent of the meteoric dust accreted by Earth is thought to be associated with present or past comets and the remainder contributed by asteroidal matter. Estimates of the flux of particles as a function of mass near

Earth, in the asteroid belt (2 to 4 AU) and near Jupiter (Ref. 1), are presently uncertain. Particles in the mass range 10^{-6} to 10^{-3} gm present the principal hazard to space vehicles and their flux in the asteroid belt is estimated to be 100 times that at Earth.

The gravitational attraction of the planets is expected to be responsible for a concentration of dust in the ecliptic plane; out-of-plane measurements will provide the first direct verification.

C. ENVIRONMENTAL PHENOMENA DURING JUPITER ENCOUNTER

The environmental phenomena of interest during the Jupiter encounter phase are

- Jupiter's magnetic field
- Trapped radiation belts
- Sources of radio-frequency emissions
- Atmospheric composition
- Atmospheric temperature profile
- Ionospheric characteristics
- Electric field characteristics

1. Radio Emissions

Earth observations of radio emissions from Jupiter can be divided into three categories, according to their frequency range and probable mechanism of production:

(1) Decameter: burst-like radio emission (mechanism uncertain)

(2) Decimeter: synchrotron radio emission

(3) Centimeter and shorter wavelength: thermal emission.

The origin of the sporadic burst-like decametric radiation is not known with certainty (Refs. 14 and 15) although its periodicity has been found to be so consistent that it has been used as a basis for a system of longitudinal coordinates. Analysis of data obtained since 1957 shows that Jupiter's satellite I_0 appears to control the rapid fluctuations of decametric emission. One mechanism which has been suggested is that I_0 moves within Jupiter's magnetosphere near the magnetic shell parameter $L = 6$ and accelerates clusters of electrons so that they move along the magnetic field lines, generating radio emissions at the local gyrofrequency. A recent explanation by Gledhill (Ref. 16) indicates a magnetic field of 30 Gauss at Jupiter's equatorial surface.

Observations of the intensity, polarization, and spatial extent of the decimeter radiation strongly suggests that it is synchrotron radiation from energetic electrons trapped in Jupiter's Van Allen belts. The spectrum in the 10 to 100 cm range is very flat with a flux density of about 6.7×10^{-26} $w/m^2/Hz$. Based on this interpretation of the decimetric radiation, models for Jupiter's magnetic field suggest a surface field strength in the range 10 to 100 Gauss, inclined at an angle of $\sim 10°$ to the rotational axis. A peak electron flux of 10^7 electrons/cm^2/sec is anticipated at 3 R_j with electron energies less than 100 MeV. Although there is no direct evidence for its existence, a proton belt with a maximum flux of 10^9 protons/cm^2/sec at 8.5 R_j with proton energies less than 4 MeV has been postulated by analogy with Earth's belts. A flyby of the planet at a radius of closest approach of 8 to 10 R_j should be capable of measuring the belt intensities directly and provide a thorough mapping of the Jovian magnetic field.

The radiation below 3-cm wavelength follows the λ^{-2} dependence of the Rayleigh-Jeans law and is primarily thermal in origin. The equivalent disk temperature $\sim 130°K$ agrees with the 8 to 14μ measurements of Wildey (Ref. 17). One of the unresolved problems concerning Jupiter is its thermal imbalance. Based on a radiant temperature of 130°K, Jupiter radiates about 2.6 times its solar input, the balance being supplied by the planet itself. Smoluchowski (Ref. 18) dismisses radioactive decay as an inadequate source and suggests that the heat may be due to a phase change of the hydrogen in the planet from molecular to metallic—a radial contraction of about 1 mm per year would be adequate. At present, there is insufficient evidence to choose between this explanation and Hubbards' hypothesis that impurities (helium) in the molecular hydrogen might have reduced the conductivity sufficiently that the excess heat is due simply to the slow cooling of the planet.

2. Atmospheric Composition

The main constituents of Jupiter's atmosphere are thought to be hydrogen, helium, methane, and ammonia. Methane and ammonia bands dominate the red end of the spectrum; however, estimates of the relative abundance of hydrogen-helium require UV measurements.

Aerobee results obtained by Stecher (Ref. 19) show a rise in albedo towards 200 Å, which is expected for a Rayleigh scattering atmosphere. Computed curves for 4.6 km-atm and 27 km-atm of hydrogen bound the experimental results and 10.2 km-atm gives the best fit. Introduction of various amounts of ammonia has been proposed to explain the flattening of the albedo below 2,300 Å but the results for the best fit (0.03 cm-atm) appear unconvincing. No satisfactory explanation has been proposed for the dip at 2,600 Å, although benzine has been suggested, as have other molecules of biological significance (Sagan Ref. 20). Jenkins (Ref. 21) suggests that the hydrogen abundance can be estimated from the reflected spectrum at 1216 Å and neighboring wavelengths corresponding to energy differences due to Ramon scattering.

D. SCIENCE EXPERIMENTS

1. Selected Representative Experiments

A comprehensive list of possible experiments, which would be appropriate for the scientific objectives outlined above, is provided in Reference 1. In order to exercise the spacecraft design, a representative science payload has been selected from this 'shopping list' of experiments. The selected scientific payload conforms to a weight limit of 50 pounds and contains instruments which, because of their susceptibility to magnetic and radiation effects, require boom mounting and preferential shielding from the RTG environment. The selected experiments and their requirements are listed in Table 5. During the interplanetary phase only the first six instruments would be operating with a measurement cycle rate of, at most, a few cycles per minute. Including engineering telemetry and reference data, the course data rate is about 32 bps. During the encounter phase, all the experiments would operate, although not simultaneously. The sample rate would increase to 10 cycles per minute and the data rate, including engineering and reference, would rise to 120 to 200 bps.

2. Imaging Experiment

Alternative methods of obtaining visual images of Jupiter during the flyby were investigated. A television camera-magnetic tape recorder system was selected since it seemed best able to provide the desired quality image with essentially space proven hardware.

The objective of the experiment was to obtain substantial area coverage of the planet with an order of magnitude improvement in resolution compared to photographs obtained from Earth. Photographic resolution obtained from Earth Telescopes is limited to approximately 1 arc-second (Ref. 22) by the atmosphere. This corresponds to a surface resolution of about 2000 km at Jupiter's opposition.

Table 5

Selected Experiments and Their Requirements

Instrument	Weight (pounds)		Power (watts)	Remarks
	Body	Remote		
Fluxgate magnetometer	3.5	1.5	4.0	Background field $\sim 0.1\gamma$
Search coil	2.0	2.0	2.0	
Plasma probe	6.0	0.0	4.0	Narrow angle ($\simeq 15°$)
Solar cosmic ray	2.0	4.0	1.5	Solar-oriented high rate, low Z
Galactic cosmic ray	3.0	2.0	2.0	Anti-solar, low-rate, include high Z
Micrometeoroid detector	5.0	0.0	1.5	
Trapped radiation	4.0	1.0	1.0	Planet oriented
IR radiometer	5.0	0.0	3.0	Planet oriented
UV H/He resonance	3.0	0.0	2.0	Planet oriented
Radio Emissions	4.0	0.0	2.0	Antenna on booms

The principal problems in obtaining 200 km resolution pictures during Jupiter flyby are the low incident light level at Jupiter and image smear caused by the spacecraft's spinning motion. For the missions considered in this report the distance of closest approach to Jupiter has been in the range of from 8 to 10 R so that a 200 km resolution element subtends approximately 1 arc-minute. In order to limit the degradation in image quality to 10 percent the linear image motion during the exposure time should be held to one-half a TV line. This can be achieved either by using a very short exposure time of approximately 0.2 msec for the 3 rpm spin rate, or providing an image motion compensation (IMC) system. A 90-percent accurate IMC system would permit relaxation of the exposure time to 2 msec. In the interests of simplicity an image motion compensation system was not considered further. The solar constant at Jupiter is approximately 450 foot-candles and the visual albedo 0.445. For f/1 optics, which may be unrealistically large for this application, and the 0.2 msec exposure time the faceplate exposure is 0.01 foot-candle-second. This is at the low end of the typical vidicon operating range of 0.003 to 0.1 foot-candle-second so that a more sensitive tube is required. With lighter weight f/3 optics, and the same 0.2 msec exposure time, the exposure is 0.001 foot-candle-second, which corresponds to an integrated light flux of 3.5×10^{-6} lumen-seconds over a typical 0.7 inch square photocathode. At this level either an intensifier vidicon or SEC vidicon comes closest to the ideal tube performance and should provide a signal-to-noise ratio of 60 to 1 (36 dB) for a 500-TV line system. The preferred solution is based on a tube of this type. The 500-TV line format provides a total field of view of four degrees and the corresponding focal length is 10 inches. Jupiter subtends an angle of about 12 degrees at closest approach and fills the field of view from a distance of 30 R_j.

Each picture would represent approximately 1.8×10^6 bits, assuming 5-bit digitization. A camera storage time of 200 seconds would lead to a readout rate of 9000 bps. The communication rate at encounter range is about 500 bps and an 18:1 read-in to read-out rate for a tape recorder is reasonable. Each picture requires one hour of transmission time, and for a series of 30 pictures, the total storage requirement is approximately 6×10^7 bits.

An appropriate picture-taking sequence depends on the flyby geometry of the particular mission. For a trailing-edge passage, Figure 27A, typical of an in-the-ecliptic mission to 10 AU, closest approach (P) occurs a few degrees beyond the terminator. Assuming that adequate illumination is available to within 20° of the terminator, the planet range will be such that a resolution of ~ 300 km (at point X) can be achieved. For loading-edge passage, Figure 27B, typical of out-of-the-ecliptic missions, closest approach occurs approximately 70° beyond the terminator and the best resolution available will be ~ 400 km at point Y.

Characteristics of the TV system are summarized in Table 6. The weight and power estimates are based on components with similar characteristics and include the optics and pointing mirror weight.

It is apparent that although TV pictures are highly desirable, both politically and scientifically — for example to investigate the Great Red Spot, the weight and power requirements to provide a really worthwhile experiment are a substantial fraction of the total science payload of a GJP class of vehicle. If required, fewer and/or lower quality pictures could be provided for a smaller weight allotment, but their value in comparison with Earth-based or possible Earth-orbiting photographs soon becomes doubtful.

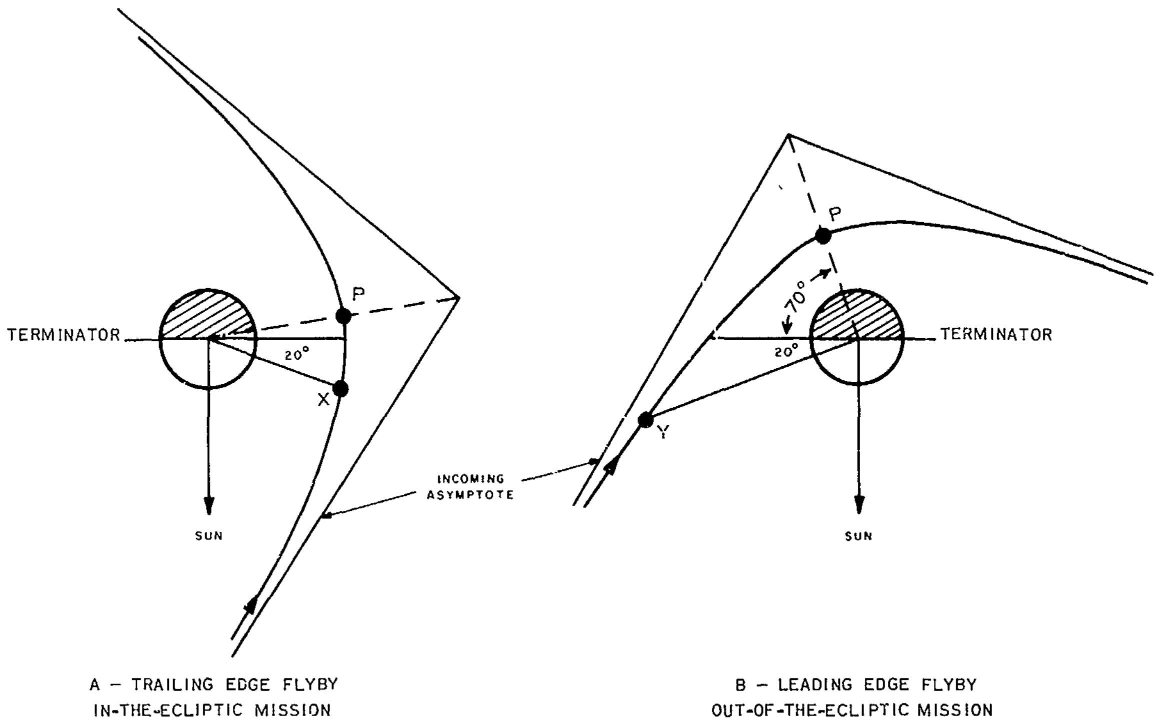

Figure 27. Typical Flyby Illumination

The spinning spacecraft poses imaging problems which may be eliminated by use of a stable platform. A concept of such a platform was prepared and is covered in Reference 37.

Table 6

Summary of a TV System Suitable for Jupiter Flyby

Parameter	Value	Notes
Camera type	1-in. intensifier vidicon or SEC vidicon	
Size Weight Power	1/6 ft^3 12 pounds 8 watts	
Sensor resolution	500 TV lines	
Angular resolution	1 arc min/line pair	
Total field-of-view	4°	
Surfacr resolution	200 km/line pair	At closest approach of 10 R_j
Coverage/frame	50,000 km^2	
Optics	10 in. focal length f/3 [e.g., Kinoptic (cine)]	
Exposure time	0.22 mseconds	
Exposure	0.001 ft-candle-second	S/N = 60:1 (35 dB)
Integrated flux at photocathode	3.5×10^{-6} lumen-second	
Picture content	1.8×10^6 bits (5-bit digitization)	
Frame time	200 seconds	
Readout rate	9 k bits/second	18:1 read-in to read-out rate
Transmission rate	500 bits/second	
Transmission time	1 hour/frame	
Tape storage req't.	6×10^7 bits for 30 frames	
Tape recorder		
Size Weight Power	1/2 ft^3 15 pounds 12 watts	

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SECTION V

SPACECRAFT SUBSYSTEM REQUIREMENTS

A. GENERAL DISCUSSION

With the identification of the GJP Phase-A Study concept (Ref. 1) as the baseline spacecraft configuration, it became appropriate to compare the subsystem requirements arising from the NEW MOONS model missions with the capabilities of the baseline spacecraft. A review of the GSFC Phase-A Study was conducted (Ref. 23), from which it was concluded that the choice of subsystems and the proposed implementation modes are valid for the two-planet and out-of-the-ecliptic missions; however, certain subsystems modifications are indicated.

The principal product of this comparison has been the definition of critical areas requiring technological advance over the GJP for the NEW MOONS missions. The areas of principal concern are as follows:

(1) The extended duration of the missions (~3 years plus for 10 AU or out-of-ecliptic missions, ~10 years for the Grand Tour) places a particular premium upon both reliability and the capability of the system to function in a reduced mode under non-catastrophic failure conditions.

The operational lifetimes required by the various missions are shown in Figure 28 with the occurrence of significant events indicated by arrowheads. Thus, a simple mission to the vicinity of Jupiter requires about 1 1/2 years; the baseline deep-space — 10 AU mission about 3 years; and an out-of-the-ecliptic mission about 3 years to reach its maximum elevation above the plane and with subsequent return to the ecliptic after 4 years and maximum southerly declination after 5 years. The Grand Tour Mission indicates approximate times of planetary encounter, Jupiter: 1 1/2 years, Saturn: 3 years, Uranus: 7 years, and Neptune: 10 1/2 years. Typical durations for Jupiter swingby to the outer planets are: Saturn: ~3 years, Uranus: ~6 years, and Neptune: ~9 years.

A review of NASA-launched spacecraft indicates that it is reasonable to postulate mission lifetimes in the 3- to 5-year range, since at least 24 spacecraft have operated continuously for from 1 to 2 years and of these 12 are still operating. Approximately twelve spacecraft have

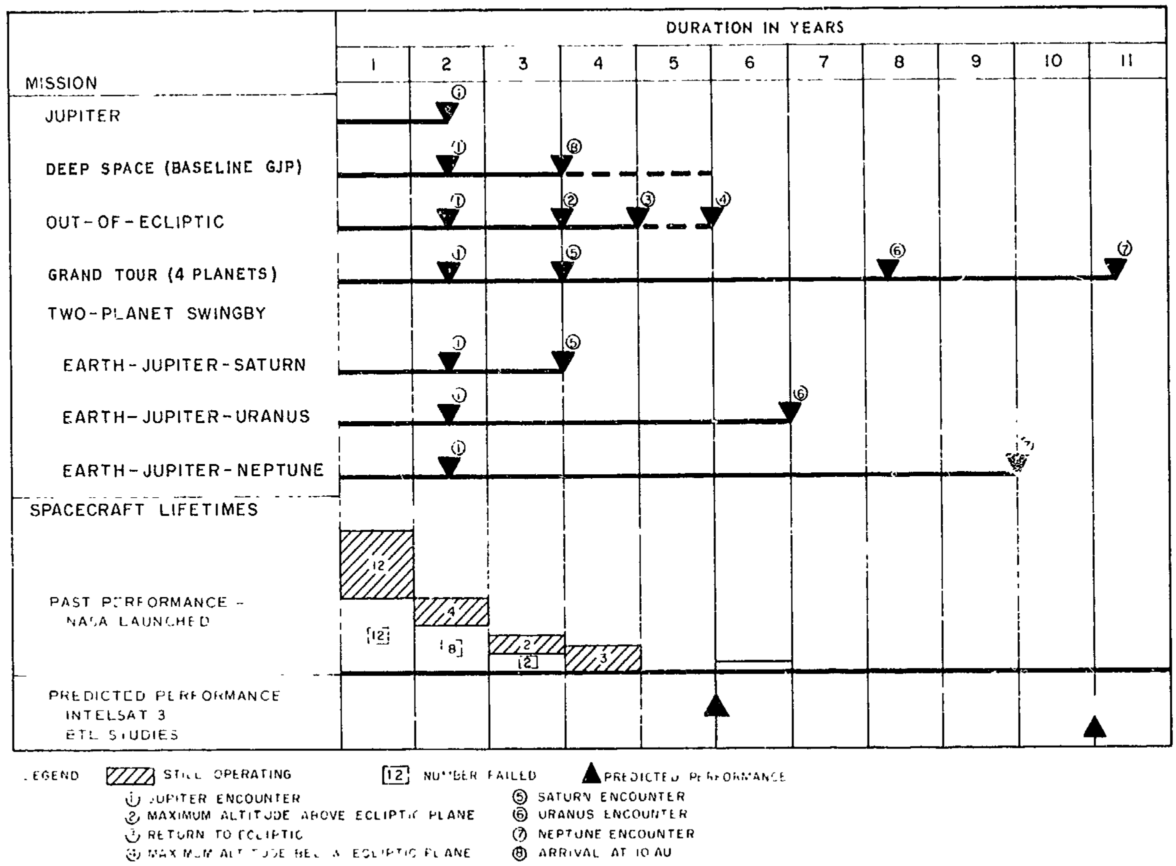

Figure 28. Mission Lifetime

operated continuously for from 1 to 3 years and of these 4 are still operating. Three spacecraft having operated for more than 4 years and continuing to do so. In addition to this prior experience, the next generation of communication satellites are required to operate for a 5-year period.

From the point of view of lifetime then, the 10 AU or out-of-ecliptic missions appear feasible against the present background of experience and design practice as do the Earth-Jupiter-Saturn swingbys. However, the Grand Tour Mission appears to require a mission lifetime beyond that which can be presently supported by experience. Some analysis by Bell Telephone Laboratories (Ref. 24), using their experience of unattended undersea-cable repeater operation, does indicate that 10-year life may not be too unreasonable.

(2) For the out-of-ecliptic missions, an arbitrary-pointing midcourse-correction capability is required. The anticipated magnitude of the

correction is similar to the GJP requirement of 100 m/sec, but to accommodate large angles between the spin axis and the ecliptic plane, both during trajectory corrections and during the post-encounter cruise, the present restricted-view Canopus sensor should be replaced by an alternative star sensor.

The two-planet swingby missions require more accurate trajectory guidance than the out-of-ecliptic mission, though the same basic system should be adequate. This involves Earth-based tracking and command guidance of the spacecraft involving two arbitrary-pointing pre-Jupiter encounter trajectory corrections and one post-encounter correction. The maximum magnitude of the ΔV requirement will be of the order of 150 m/sec, assuming use of the unguided TE 364-4 as the upper stage of the launch vehicle. The maximum ΔV will be less than 50 m/sec if a guided upper stage, such as Burner II, is used in conjunction with a more energetic SLV3X first-stage booster.

For the Grand-Tour Mission, multiple arbitrary-pointing trajectory-correction maneuvers will be required. The magnitude of the corrections will be considerably in excess of the baseline 100 m/sec, and the spatial definition of post-encounter corrections poses severe problems. The thermal and guidance problems are of different magnitudes. A remedy for the thermal problem is clearly in sight, all that is necessary is to supply a sufficient (thermal) power margin. This can be done by several techniques such as either directly dumping waste RTG heat into the spacecraft or via electrical power generation. The guidance problem is more significant in that the trajectory data of interest is clearly obtainable in principle but may presently be out of reach in practice.

B. SUBSYSTEM CONCEPTS

The spacecraft makeup for the precursor missions of interest may be defined with the aid of Table 7.

1. Communications

The need for good communication capability at the time of Jupiter encounter and for adequate capability at extreme ranges, yet with only modest on-board power, sets a requirement for high antenna gain. The baseline GJP system provides a maximum down-link telemetry data rate of over 800 bps at Jupiter encounter (~4.2 AU), and with the omni-antenna command up-link reception at

Table 7

Comparison of Subsystems for NEW MOONS Missions

Subsystem	Baseline GJP Capability (Ref. 1)	Out-of-Ecliptic Reqm'ts	Two-Planet (Jupiter-Saturn) Swingby	Grand Tour Requirements
Communication	1) 9-ft diameter fixed dish 2) 10-watt S.S. transmitter 3) 210-ft DSN - 830 bps at 4.2 AU 4) 85-ft - 160 bps at 3.5 AU	Satisfactory Satisfactory Satisfactory Satisfactory	Satisfactory Satisfactory 160 bps at 10 AU —	Unfurlable - 30 ft Satisfactory 100 bps to 30 AU —
Data Storage	1) Plated wire store 2) 450,000-bit capacity	Adequate, but not preferred Satisfactory	Adequate Satisfactory	 Larger Capacity
Attitude Control	1) Spin stabilized 2) Earth pointing ~1° 3) RF sensing 4) Sun and Canopus reference 5) Cold gas torquing	Satisfactory Satisfactory Satisfactory Sun and star reference Requires additional gas	Satisfactory Satisfactory Satisfactory Sun and star reference Requires additional gas	Satisfactory 0.3 deg. Incomplete analysis Planetary approach guidance As yet unspecified
Trajectory Correction	1) $\Delta V \approx 100$ m/sec 2) Hydrazine 3) Multiple start	Satisfactory Satisfactory Single arbitrary pointing	$\Delta V \approx 150$ m/sec* Satisfactory 3-arbitrary pointing	$\Delta V > 100$ m/sec — Multiple arbitrary pointing
Thermal Control	1) 1 to 10 AU solar range 2) Active control	1 to 5 AU solar range Satisfactory	1 to 10 AU solar range Satisfactory	1 to 30 AU solar range Satisfactory
Power	1) 2 × 75 watt (E) RTG 2) PbTe SNAP-27 type	Satisfactory SiGe reduces magnetic contamination	Satisfactory SiGe reduces magnetic contamination	Additional RTG's SiGe reduces magnetic contamination
Booster	SLV3C/Centaur/TE 364-4	Satisfactory	SLV3X/Centaur/TE 364-4 or TitanIIID/Centaur	Titan IIID/Centaur

*For the Titan IIID/Centaur booster the baseline ΔV capacity is satisfactory

1 bps out to 6 AU. [illegible] ranging with 50 meters resolution is provided out to 5 AU, using the high-gain spacecraft antenna; Doppler information can be obtained to distances of about 7 AU with the omni-antenna.

This telemetry performance is provided by a fixed 9-ft. dish and a 10-watt (2 × 5 watt) solid-state transmitter, using convolution coding and sequential decoding. The system is configured to operate with the 210-ft. DSN network for encounter bit rates of 830 bps and an improved 85-ft. STADAN network for cruise and lower-bit-rate modes. The command link performance is provided by redundant phase-locked receivers that may be switched to either omni or high-gain antenna. PM/FSK modulation is used at a command rate of 1 or 10 bps. Both discrete and quantitative commands are used, with provisions for storage of commands for later execution.

An on-board data-handling system provides the signal conditioning and coding for the experiments. The use of a central data processor is considered to perform scaling, compression, integration, and comparison of the experimental data. A plated wire, with a bulk-storage capacity of 450,000 bits is sized to permit data storage at 28 bits per second for four hours.

Actual data requirements for the out-of-the-ecliptic missions are expected to be considerably less than the baseline capability. In fact, a good case can be made for an encounter data rate in the range of 100 to 200 bps and a cruise data rate of approximately 30 bps. The baseline system can, therefore, support both the out-of-the-ecliptic mission and the Jupiter-Saturn swingby using the 210-foot dish for encounter data and the 85-foot dish for cruise data.

Higher bit rates at encounter are useful but do not radically change the system capability unless they become very much higher. If imaging data is required, then the proper course appears to be to provide adequate data storage, as proposed in Section IV D2, rather than attempt real-time transmission.

The proposed GJP system provides a capability of 160 bps to 10 AU, which is adequate for the Jupiter swingby to Saturn, and a very low rate of approximately 1 bps out to 30 AU with the same 9-foot antenna and transmitter power. A Grand Tour to Neptune would not be adequately supported and, clearly, if the RF power cannot be increased, then the antenna gain has to be. An unfurlable antenna, such as the Application Technology Satellite (ATS) design of 30 ft. diameter, could raise the capability to approximately 100 bps at 30 AU.

2. Attitude Control

For the GJP mission, a fixed antenna having the maximum diameter allowed by the booster shroud (9 ft.) was postulated in the interest of reliability.

This high-gain antenna inevitably is of narrow beam width ($\sim 3°$), and since spin stabilization is fundamental to an early, long-lived mission, the cruise geometry emerges as Earth pointing with an accuracy of $\sim 1°$. With an Earth-pointing spin-stabilized vehicle carrying an on-axis high-gain antenna, a number of possibilities exist for jet attitude control. This function can be performed by a cold gas system that avoids undesirable combustion products which could interfere with the spacecraft scientific experiments. For the Grand Tour Mission using a 30-foot dish, the Earth-pointing accuracy should be improved to 0.3 degree.

The attitude-sensing problem has several possible, but not necessarily exclusive, solutions. If only Earth-directed orientation is required then the spacecraft's RF system based upon the up-link data is acceptable. The use of a Sun-star attitude-sensing system allows for arbitrary pointing and for on-board closed-loop pointing control. There are strong arguments for inclusion of both Earth-line sensing and celestial sensing which is the course taken by the GJP. However, the Sun-Canopus system obviously can only work while the appropriate sensor has a view of Canopus. As proposed in the Phase A Study Report, this sensor is masked off at $\pm 20°$ and since Canopus lies $15°$ off the ecliptic pole, the actual free view is only $\pm 5°$. Closed-loop control is then only possible to this angle out-of-the-ecliptic.

3. Trajectory Correction

Analysis indicates that a trajectory correction scheme restricted to an Earth-pointing propulsion capability imposes only modest penalties upon simple swingby missions (Ref. 1) but cannot meet the requirements for out-of-the-ecliptic missions. This latter requires an unfocused impact zone to lie roughly in a 75,000 km radius circle (Figure 16). Since an uncorrected trajectory can be expected to fall within an ellipse measuring 2,500,000 km, see Table 3, in azimuth by 600,000 km in elevation (at the 1σ level) it emerges that only azimuthal correction is required for an in-plane flyby, but an out-of-the-ecliptic mission requires arbitrary pointing.

A single, arbitrary, midcourse correction applied some ten days after launch can reduce both components of the miss to about 25,000 km (1σ), which is adequate for the out-of-ecliptic missions. A second arbitrary maneuver some 100 days after launch can virtually eliminate all but trajectory determination errors. Accuracy of this order is necessary for the economical implementation of two-planet swingby missions, which, in any event, require an arbitrary correction maneuver following flyby. The Grand Tour trajectory-correction requirements are impossible to specify precisely at this time.

In summation, it is apparent that all but the simplest missions benefit from the capability for arbitrary propulsive maneuvers and that this capability becomes essential for multiple-planet swingby missions. The use of a Sun-star attitude sensing system allows for on-board closed-loop arbitrary pointing, although of course, complications arise if substantial out-of-the-ecliptic angles become operationally necessary. Open-loop ground commanded control is a possibility under conditions of extreme demand.

The choice of a propulsion system is almost self-defining in that the payload weight for the out-of-the-ecliptic and two-planet swingby missions is by definition small (550-600 pounds), as is the velocity correction requirement, ($\Delta V \approx$ 100 cm/sec). The total impulse demand (approximately 6000 lb-sec) is such that the use of a monopropellant hydrazine system clearly offers the best combination of simplicity and low weight—and in fact would be the proper selection even for a significantly increased demand. The long mission durations that are proposed favor the selection of a simple system, even at some cost in weight.

4. Thermal Control

The NEW MOONS missions have in common a difficult thermal environment. The most severe is the multiple-planet swingby, where the probe will move out to 30 AU compared with the 5 to 6 AU maximum solar range of the out-of-the-ecliptic mission. In order to cover the case of a simple swingby, analysis has been performed for a nominal range of 10 AU in Task V, Spacecraft Analysis and Design. The critical conclusion of the analysis, reported in detail in Task V, is that if thermal "lumping" of the power dissipating equipment in the main body of the spacecraft can be achieved then a satisfactory thermal design can be developed. The required internal dissipation to achieve the desired temperature range of 0° to 40°C, including gradients, is highly related to the actual equipment layout. For the design analysis a nominal 75 watts was used and the temperature for the operating equipment was within the desired range. Consideration has been given to the possibilities of an electrical failure of one of the two RTG's carried by the GJP, and it is apparent that such a failure reduces the mission capability at large solar ranges but still permits spacecraft operation on a partial mode.

5. Data Storage

The question of data storage capability has been discussed briefly. It appears that even the simplest model mission could benefit from the provision of at least moderate data storage (say 60 minutes of data at 150 bps for a minimum of 500,000 bits) while a more advanced mission would require far more capacity. An estimate of the data storage requirement for a series of television pictures of Jupiter is about 6×10^7 bits which could be met by a magnetic tape recorder. Reserve capability might be provided by the plated wire store suggested by GJP, although this seems unduly heavy for its rather limited capacity.

6. Power System

The mission duration for the out-of-ecliptic mission is approximately 3 years (to point of maximum elevation above the ecliptic plane) as is the Jupiter swingby to Saturn, and the power-demand profiles are essentially unchanged from the baseline GJP mission. An end-of-life requirement for 100 watt (e) source is adequate for both out-of-ecliptic and two-planet swingby missions. At Jupiter encounter, at a solar range of approximately 5 AU, and assuming that conventional N/P solar cells are capable of producing 0.4 watt/ft^2, then, a 100-watt system would require a 250 ft^2 array. Using conventional array structure, the weight would be approximately 250 pounds. The weight penalty compared to an RTG system is about 150 pounds at Jupiter and becomes rapidly worse with increasing distance from the Sun (Ref. 2). For Jupiter-swingby missions to 10 AU to Saturn or AU to Saturn or for a Grand Tour, the weight of solar-cell systems is completely prohibitive. For out-of-the-ecliptic mission of 5 AU and less a sun orienting system must be added to the array which will have the effect of only a small increase in array weight but with a significant increase in system complexity.

A Grand-Tour Mission would have a duration of > 10 years, therefore, if 100 watt (e) end-of-life capability is still considered adequate the beginning-of-life capability would have to be approximately 300 watt (e), which is roughly double the GJP value (assuming ~ 8 percent per year degradation). Thus, four RTG units would be required as compared with two for the baseline GJP and the out-of-ecliptic missions.

A modified version of the SNAP-27 has been proposed as a suitable power source for the baseline GJP. For follow-on missions, such as considered in this report, substitution of a system using SiGe thermoelectric elements instead of lead telluride should be considered for reducing magnetic contamination.

Magnetic tests of a SNAP-27 generator at GSFC (Ref. 2) have shown that the iron in the hot shoes associated with the PbTe elements plus the stray field create a relatively large magnetic field. To reduce the RTG-produced background magnetic field at the magnetometer to a tolerable level of 0.1 gamma would require an RTG-sensor separation of about 6.3 meters. No magnetic materials are required with the SiGe elements and the only source of magnetic fields are current loops within the RTG. Generally, these are relatively easy to compensate by careful circuit design so that the residual dipole moments are very small. For more detailed discussion on this subject see Task III, Techniques for Achieving Magnetic Cleanliness.

A PbTe RTG operates in a sealed inert-gas environment, since the material oxidizes in air and sublimes in a vacuum at the present SNAP 27 operating temperatures. The SiGe thermocouples, on the other hand, can operate either in air or in vacuum. If leakage in the PbTe containment system develops over the long operational life of the generator, or by postulating a meteoroid puncture in the containment systems, this could result in a substantial power reduction with time.

A more detailed discussion on the relative merits on all candidate RTG technology can be found in Reference 2.

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SECTION VI

CONCLUSIONS AND RECOMMENDATIONS

From the analysis of Sections III through V it is apparent that with comparatively minor modification the GJP spacecraft concept is capable of performing a scientifically attractive out-of-the-ecliptic mission in addition to its intended deep-space in-ecliptic mission. In particular it should be noted that:

1. An Atlas SLV3C/Centaur/TE 364-4 is capable of launching the basic 550- to 600-lb spacecraft, on a short, nominally 550-day flight to Jupiter during the 1972 and 1974 launch opportunities. Such trajectories provide near-minimum communication distances at planetary encounter and suitable arrival conditions for significant ($> 40°$) post-encounter inclinations to the ecliptic.

2. Aiming zones at Jupiter have been identified that provide excellent opportunities for planetary investigation during flyby and that lead to post-encounter trajectories which reach more than 1 AU above the ecliptic plane in the neighborhood of Earth's orbit.

   A single, arbitrary pointing, midcourse maneuver, applied some ten days after launch, with a maximum ΔV of less than 100 m/sec is sufficient to adjust the aiming point at Jupiter to lie within a 75,000-km radius circle (3σ), which is appropriate to out-of-the ecliptic requirements.

3. The principal change required in the baseline GJP subsystems is provision of a closed-loop attitude-control system capable of operating at large angles to the ecliptic plane. As presently configured, the $\pm$ 20° field-of-view of the Canopus sensor is not capable of working at more than a few degrees out of the ecliptic. During the cruise mode, at any angle to the ecliptic, the Earth-pointing spin axis could be maintained using RF information only, and an arbitrary orientation maneuver for trajectory correction could be achieved in an open-loop mode. However, in the interest of reliability it would be preferable to maintain a closed-loop capability by implementing an alternative celestial reference system. The choice of a stellar reference and sensor system compatible with the out-of-ecliptic mission requires further study.

4. Compared with the GJP mission, a multiple-planet swingby, or Grand Tour, imposes much more severe requirements on the communications systems, due to the increase in range to $\sim$ 30 AU, and, more critically, on the guidance-accuracy requirements. The thermal system appears to be adequate for the $\sim$ 30 AU range. The communication problems are amenable to conventional solutions. Thus, the communication capability can be upgraded by a combination of increased antenna gain (by means of an unfurlable antenna in place of the baseline 9-ft. fixed dish) and transmitter power. If the antenna diameter is increased to 30 feet, then the present 10-watt transmitter power can support a 100-bps rate to a 210-foot Earth antenna from 30 AU.

   A solution to the guidance problem, on the other hand, is not so straightforward. For a secondary target planet, e.g., a Jupiter swingby to Saturn, the accuracy requirements at Jupiter are an order of magnitude more stringent than the requirements for an out-of-the-ecliptic mission. If the impact parameter at Jupiter is controlled to the limit of Earth-based orbit-determination accuracy, but no post-encounter corrections are applied, the corresponding uncertainty in Saturn-flyby distance can amount to tens of planetary radii. Both pre- and post-encounter trajectory corrections are required to perform any but the coarsest two-plant flybys.

5. A Jupiter swingby to Saturn, with a flyby accuracy of approximately half a planetary radius, seems possible using Earth-based tracking and two pre-encounter and one post-encounter trajectory corrections. Such a mission is a reasonable next step after the out-of-ecliptic mission. It makes use of the arbitrary pointing capability required for the out-of-ecliptic mission and, although requiring three firings, the total ΔV required is only about one and a half times that of the GJP.

   Alternatively, use of a launch vehicle with a guided final stage, such as SLV3X/Centaur/Burner II or the Titan IIID/Centaur, reduces the injection errors to the extent that the on-board trajectory-correction requirements are within the present GJP capabilities. In addition to providing improved injection accuracy, use of these boosters extends the possible missions through the 1979 and 1980 Earth-Jupiter-Saturn opportunities. Further opportunities such as Earth-Jupiter-Uranus or Earth-Jupiter-Neptune during 1978-1982 and Earth-Saturn-Uranus or Earth-Saturn-Neptune flights through the 1980's offer attractive growth prospects for the basic GJP concept. More detailed analysis of the Jupiter-Saturn swingbys should be undertaken to define the required trajectory and to reconfirm the applicability of the basic

GJP subsystem capabilities. In addition these studies should determine the feasibility of extending the same techniques to the more ambitious two-planet swingbys mentioned above.

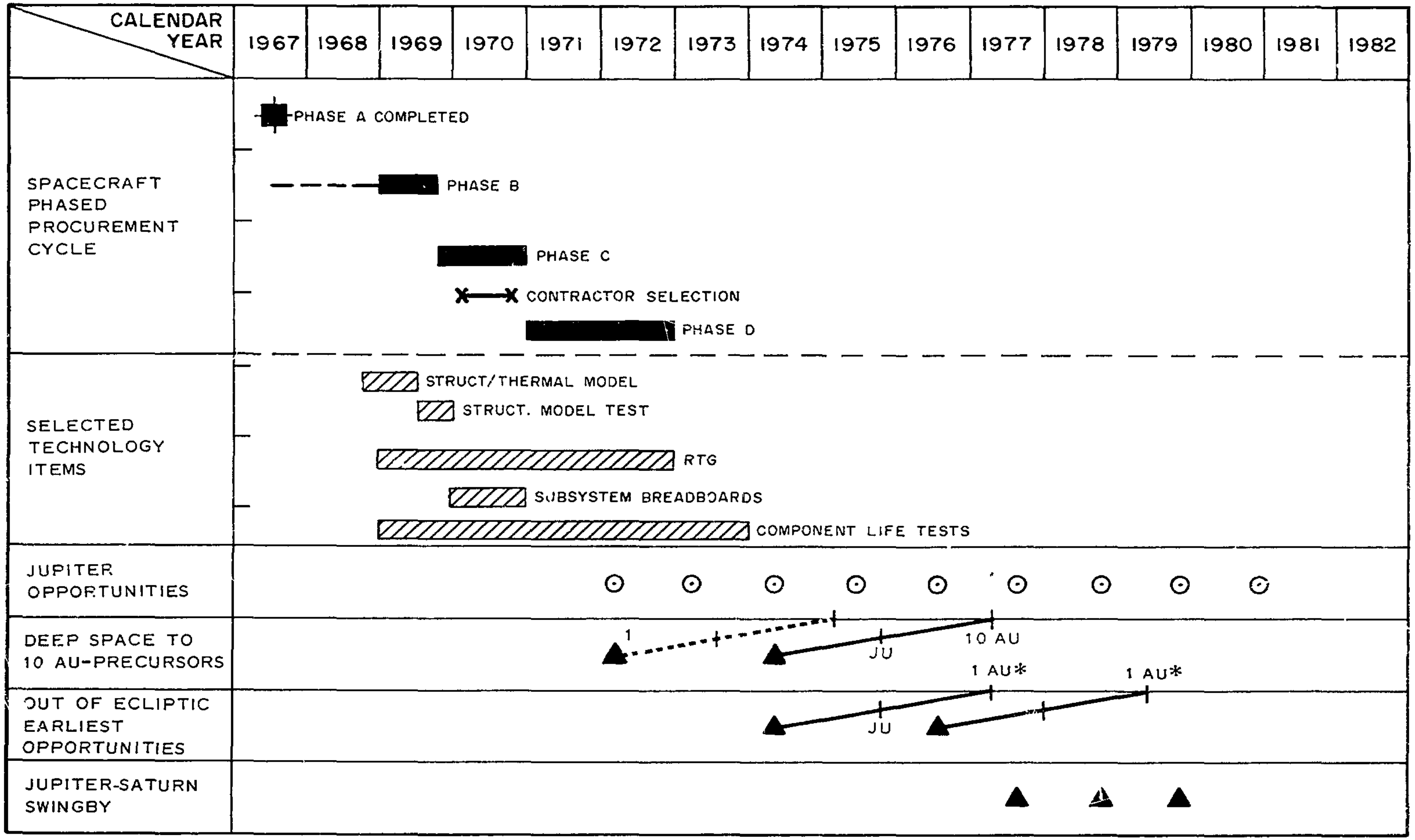
6. The weight of a solar-cell system to provide 100 watt (e) at Jupiter is about 150 pounds more than that of an RTG system. For a spacecraft with a total weight allowance in the range of 550 to 600 pounds, it is obvious that the RTG is the only reasonable power source. For missions to 10 AU and beyond, the advantages of the RTG power source are even more pronounced.

7. Two RTG units of the SNAP 27 class to provide a total of 100 watt (e) end of life ($\sim$ 5 years after fueling) are required for the out-of-the-ecliptic mission and the Jupiter-Saturn swingby, both of which have similar power profiles to the baseline GJP system.

8. A development plan is shown in Figure 29 with the object of preparing for a launch during the 1974 opportunity. The Phase-A Study performed by GSFC is taken as the basis for this family of missions, beginning with a Jupiter swingby to 10 AU, then the successively more difficult out-of-ecliptic mission, and last, the Jupiter-Saturn swingby. Phase B occupies the first nine months of calendar year 1969 and is followed immediately by a fifteen-month Phase C. During this phase, which is seen as an in-house GSFC effort with increasing contractor support, a selection of the prime contractor for Phase D will be made. Phase D will require two years for hardware development and test of the flight spacecraft.

   Key technology items that require early action are identified. (1) The RTG power-supply system should be started in 1969 to meet a four-year delivery cycle, based on SNAP-27 experience. (2) A structural and thermal model of the spacecraft can be built from the drawings available at the completion of the NEW MOONS Study and subsequently tested during 1969. (3) Guidance and control, and communication systems should be carried through detail design, breadboard, and development tests in 1969 and 1970.

   Available launch dates to Jupiter are shown as circles, and suggested missions are indicated by arrowheads. It is worthy of note that the development plan as shown cannot meet the 1972 launch date, which was particularly attractive for a deep-space mission, since its duration coincided with a period of low solar activity. Results from a launch in 1974, the earliest practicable launch date, based on assumptions



[1]NOTE: • BASED ON ASSUMPTIONS SHOWN ON THIS CHART. THE 1972 LAUNCH DATE DOES NOT APPEAR FEASIBLE PROGRAMMATICALLY

• SEE APPENDIX IV FOR OTHER LAUNCH PLANS

▲ SUGGESTED MISSIONS

* DISTANCE FROM THE ECLIPTIC PLANE

Figure 29. Development Plan for Deep Space Missions

made, will not be available in time to affect materially the design of a spacecraft for a 1976 out-of-ecliptic mission, though better planetary data will aid in trajectory-design and guidance calculations. Both the deep-space mission and the out-of-ecliptic design would contribute significantly to a two-planet swingby launched in 1977 or 1978.

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APPENDIX I

CONIC TRAJECTORY PROGRAM

The trajectory of a spacecraft on a Jupiter swingby mission can be divided into four phases of two-body motion:

(1) Earth escape phase,

(2) Heliocentric transfer from Earth to Jupiter,

(3) Hyperbolic flyby of Jupiter, and

(4) Post-encounter heliocentric phase.

The overall flow diagram of a computer program to investigate such trajectories, as shown in Figure I-1, gives the main subroutines used and the method in which they are linked together.

1. The input quantities to the program are the launch date (T_0), the time of flight ($T_3 - T_0$), and the components of the impact parameter at Jupiter $\bar{B} \cdot \hat{T}$ and $\bar{B} \cdot \hat{R}$.

2. The position and velocity of Earth at launch date (XRTH, VRTH) and the position and velocity of Jupiter at the arrival date (XPL, VPL) are obtained from an analytic ephemeris routine.

3. These initial and final positions together with the time of flight are sufficient to uniquely determine the spacecraft's heliocentric trajectory parameters. The Lambert routine determines the 'initial,' V_0, and 'final,' V_3, velocities of the spacecraft on a trajectory from the center of Earth to the center of Jupiter.

4. The conditions at the center of Jupiter are used as inputs to the TWO BODY subroutine which is an adaptation of Goodyear's equations (Ref. 23). Given the position and velocity of the spacecraft at time T_3 and the radius of the sphere of influence of the planet (R_s) the program iterates to find the time at which the spacecraft entered the sphere of influence (T_1) and its position and velocity at that time ($\bar{X}_1$, $\bar{V}_1$). The vehicle's velocity with respect to the Sun and the planet's velocity at the time of entry (obtained from the ephemeris) are used to calculate the vehicle velocity with respect to Jupiter at entry into Jupiter's sphere ($\bar{V}_1'$).

READ
TF = time of flight LD = launch date B · T = } miss vectors B · R = }
EPHEMERIS
XRTH = position of earth at launch date VRTH = velocity of earth at launch date XPL = position of planet at arrival date VPL = velocity of planet at arrival date
LAMBERT
V_0 = velocity of probe at center of Earth V3 = velocity of probe at center of planet
TWO-BODY
X2 = position of probe at edge of planet's sphere V2 = velocity of probe at edge of planet's sphere
SWING-BY
calculate and print parameters inside planet's sphere of influence
TWO-BODY
X1 = position of probe at edge of Earth's sphere V1 = velocity of probe at edge of Earth's sphere
EARTH
calculate and print parameters inside Earth's sphere of influence

Figure I-1. Overall Flow Diagram

Together with the components of the impact parameter $\bar{B} \cdot \hat{T}$ and $\bar{B} \cdot \hat{R}$ this velocity is sufficient to define the parameters of the hyperbolic flyby.

5. At entry to Jupiter's sphere of influence we print:

Vehicle velocity with respect to the Sun $\quad \bar{V}_1 \quad$ (ecliptic)

Vehicle velocity with respect to Jupiter $\bar{V}_1'$ (ecliptic)

Planet velocity $\bar{V}_{P1}$ (ecliptic)

$\bar{B} \cdot \hat{T}$ and $\bar{B} \cdot \hat{R}$

Vehicle position with respect to the planet X1P (ecliptic)

Inside the sphere of influence the SWINGBY routine calculates the parameters of the planetocentric flyby hyperbola and prints:

Semi major axis A (km)

$$A = \frac{\mu_J R_s}{2\mu_J R_s (V_1')^2}$$

Eccentricity ECC

$$ECC = \sqrt{1 + B^2/A^2}$$

Closest approach (km)

$$R_0 = A(1 - ECC)$$

Semi latus rectum (km)

$$p = A(1 - ECC^2)$$

True anomaly at entry (degs)

$$ANOM = \cos^{-1} \frac{P - R_s}{R_s \times ECC}$$

At exit from Jupiter's sphere of influence we print

Position of vehicle wrt Jupiter X2P (km)

Position of vehicle wrt Sun X2 (AU)

Vehicle velocity wrt Jupiter V2P

Vehicle velocity wrt Sun V2

Planet velocity VP

Planet position XP

Using these exit conditions the parameters of the post-encounter heliocentric trajectory are calculated; those printed are

Angular momentum

$$\bar{H}_2 = \bar{X}_2 \times \bar{V}_2$$

Orbit inclination to ecliptic (degs)

$$i = \cos^{-1}\left(\frac{\bar{H}_2 \cdot \hat{k}}{H_2}\right)$$

Semi major axis (AU)

$$A_2 = \frac{\mu_s X_2}{2\mu_s - X_2 (V_2)^2}$$

Semi latus rectum

$$P_2 = \frac{H_2^2}{\mu_s}$$

Eccentricity

$$ECC = \sqrt{1 - P_2/A_2}$$

Perihelion

$$RP = A_2(1 - ECC)$$

Aphelion

$$RA = A_2(1 + ECC)$$

6. The TWO BODY routine is again used to obtain the state vector at exit from Earth's sphere of influence, radius R_E, from the initial conditions of the Lambert solution.

The velocity of the spacecraft with respect to the Sun at exit from the sphere and the Earth's velocity are printed. The hyperbolic excess velocity at Earth departure is given by

$$\bar{V}_h = \bar{V}_{sat} - \bar{V}_{Earth}$$

Given this velocity, an Earth parking orbit radius R_P corresponding to an altitude of 100 n.mi. and a launch site at 28.5°N, the parameters of the Earth departure phase are calculated. Those printed are

Semi major axis (km)

$$A = \frac{\mu_E R_E}{2\mu_E - R_E V_h^2}$$

Eccentricity

$$ECC = \frac{A - R_P}{A}$$

Impact parameter (km)

$$B = A\sqrt{E^2 - 1}$$

Injection velocity, (km/sec)

$$V = \sqrt{V_h^2 + \frac{2\mu E}{R_p}}$$

Injection ΔV, (km/sec)

$$\text{DELTA V} = V - \sqrt{\mu E/R_p}$$

Inclination

$$\cos I = \cos \phi_L \sin \xi_L$$

where ξ_L is launch azimuth and ϕ_L is launch-site declination. Unit vectors $\hat{S}$, $\hat{W}$, $\hat{B}$ are also calculated where S is along the departure asymptote, i.e.,

$$\hat{S} = \frac{\overline{V}_h}{V_h},$$

$\hat{W}$ is normal to the geocentric orbit plane and

$$\hat{B} = \hat{S} \times \hat{W}.$$

EXAMPLE

A typical printout for a launch date of May 20, 1974 and a 550-day flight to Jupiter is provided for illustration. In this example, the components of the impact parameter are

$$B \cdot T = -1.312 \times 10^6 \text{ km}$$

$$B \cdot R = 0.707 + 10^6 \text{ km}$$

leading to a post-encounter inclination of 41.1°.

1. ENTER SPHERE OF INFLUENCE OF JUP

VEHICLE VEL, SUN	0.19906235	0.37259960	0.00079161	EMOS
JUP	9.99344635	-1.97556782	-0.00386164	KM/SEC
PLANET VEL	-0.13673902	0.43898296	0.00092137	EMOS

BDOTT = -0.13120000E 07 BDOTR = 0.70700000E 06

X1P = -0.47008096E 08 0.10630250E 08 0.68873700E 06

2. PARAMETERS OF JUPITER CENTERED HYPERBOLA

A = -0.12861050E 07	ECC = 1.53064156	RO = 0.68246075E 06
P = 0.17270620E 07	ANOM = 129.04356384	
S = 0.98101473E 00	-0.19393319E 00	-0.37908065E 03
T = -0.19393325E 00	-0.98101515E 00	0.0
R = -0.37188386E 03	0.73516334E 04	-0.10000000E 01
B = 0.17054695E 00	0.86364150E 00	-0.47438008E 00

3. LEAVE SPHERE OF INFLUENCE OF JUP

X2P = 0.27209900E 06	-0.42624544E 08	0.22501440E 08		
X2 = 0.43875866E 01	0.20397644E 01	0.40505111E 01		
VEHICLE VEL, JUP	-0.25615239	-8.99218273	4.77984715	KM/SEC
SUN	-0.21856391	0.10614985	0.16332948	EMOS
PLANET VEL	-0.20995665	0.40830654	0.00271636	.EMOS
POS	4.38576794	2.32468796	-0.10990554	AU

4. POST ENCOUNTER TRAJECTORY PARAMETERS

ANGULAR MOMENTUM	0.32885396	-0.72547513	0.91156048.
INCL = 41.14718628	A2 = 3.15352345	P2 = 1.46540070	
ECC = 0.73165107	RP = 0.84624463	RA = 5.46079922	

5. AT EARTH'S SPHERE OF INFLUENCE

EXIT VELOCITY	1.09160900	-0.64712405	-0.33589947	EMOS
EARTH VELOCITY	0.82931578	-0.49156237	-0.21314055	EMOS
HYPERBOLIC EXCESS	7.80584431	-4.62951469	-3.65330410	KM/SEC

6. EARTH-CENTERED DEPARTURE HYPERBOLA

A = -0.42023359E 04	ECC = 2.56119251	B = -0.99086953E 04
V = 0.14738483E 02	DELTA V = 0.69438436E 01	
COSI = 0.87881726		

6. EARTH-CENTERED DEPARTURE HYPERBOLA (continued)

S	=	0.79788572E 00	-0.47321260E 00	-0.37342781E 00
W	=	0.46761674E 00	0.94946563E 01	0.87881726E 00
B	=	-0.38041168E 00	-0.87581676E 00	0.29703861E 00
S	=	0.79788572E 00	-0.47321260E 00	-0.37342781E 00
W	=	0.14093703E 00	0.45586956E 00	0.87881726E 00
B	=	-0.58610171E 00	-0.75382549E 00	-0.29703867E 00

APPENDIX II

N-BODY TRAJECTORY PROGRAM:

GRAVITY ASSISTED SPACE PROBE (GASP)*

A. INTRODUCTION TO THE PROBLEM

For this study a special starter was added to the ITEM program. This starter finds the initial conditions for an integrated trajectory to a specified planet, when given a starting Julian Date and a desired flight time in days. These conditions (on an INPUT control) are either at the Earth's sphere of influence (in Sun reference) or on a specified circular orbit around the Earth (in Earth reference). When the option to find the conditions on the parking orbit is used, the starter also finds the time of launch necessary to achieve this trajectory without a dog leg, the position on the parking orbit at burnout, the launch azimuth, the pitch and yaw angles used for leaving the parking orbit, and the burnout payload needed to attain a final weight (INPUT).

The following quantities are inputs to the program:

1. Starting Julian Date.

2. Desired flight time in days.

3. A specified target planet.

4. An offset in days for the position of the target planet at arrival time.

5. A specified distance in kilometers for positioning the target out of the ecliptic plane.

6. An option (a) for starting the trajectory from the Earth's sphere of influence or (b) from a circular parking orbit of specified radius.

7. If option (b) is used, the following has to be supplied: location of launch site, pounds of force for two burns used to leave parking orbit, IPS's, final weight desired, weight to be dropped after first burn, time in hours to reach the parking orbit from a central ascent angle between a station and the burnout point on the parking orbit.

*This Appendix is excerpt from Final Report, "Gravity Assisted Space Probe (GASP)," Pines, S., and Lefton, L., Report No. 68-11, Analytical Mechanics Associates, Inc., (May 1968).

In addition, special output options were provided, as explained below:

1. For OPTION (a) the integration starts at the Earth's sphere of influence in Sun reference and continues until a specified maximum time is reached, printing at specified intervals.

2. For OPTION (b) the program prints the above-mentioned information before starting the trajectory.

In both cases the printouts may be the normal ITEM output; however, under special control, this output is in ecliptic coordinates and:

1. In Earth reference, positions are in kilometers and velocities in kilometers per second.

2. In Sun reference, positions are in AU and velocities in kilometers per second.

3. In Jupiter reference, positions are in Jupiter radii and velocities are in kilometers per second.

4. Spin axis — Sun line angle in degrees.

5. Spin axis — Earth line angle in degrees.

6. Spin axis — Jupiter line angle in degrees.

7. Earth — Vehicle — Sun line angle in degrees.

8. If desired, any number (30 maximum) of radar station observations are printed giving the following: L, M, azimuth, elevation, topocentric right ascension, topocentric declination, slant range, and range rate.

9. Upon entering the target's sphere of influence the impact plane parameters, including $B \cdot \hat{T}$ and $B \cdot \hat{R}$, are printed.

B. METHOD OF SOLUTION

The positions and velocities of the Earth (R_e and $\dot{R}_e$, heliocentric) are found for the starting Julian Date (T) by looking up an ephemeris. Similarly, the positions and velocities of the target planet (R_T and $\dot{R}_T$, heliocentric) are found for the starting time plus time-of-flight ($T + \Delta T$). Now Lambert's problem is solved to determine the conic which will proceed from the Earth to the target in ΔT days, by finding the velocity vector at the Earth. Next, the time spent traveling on this conic from the Earth to the Earth's sphere of influence is found, and a two-body solution is used to determine the position and velocity vectors at that point. A flow chart for the Lambert option is shown in Figure II-1.

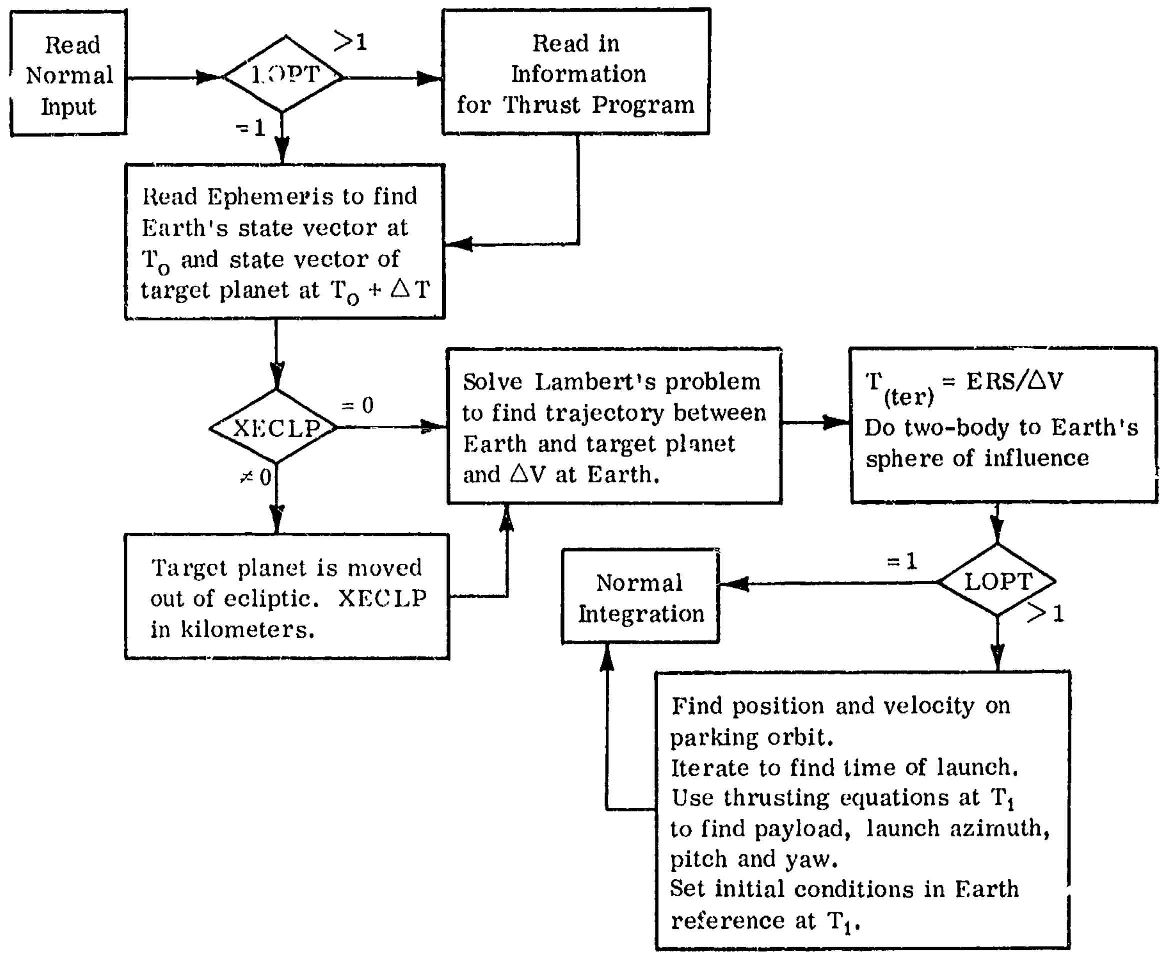

Figure II-1. Flow Chart for Lambert Option

Under OPTION (a) the main program is entered at this point. For OPTION (b), the state vector is switched to EARTH reference and we determine the conic and the time transpired in traveling to the sphere of influence from a point on a parking orbit ($T_{ER} - T_1$). Now the launch time (T_0) is found, such that the vehicle can take off from the pad, achieve a parking coast on this orbit to (T_1), the time the vehicle leaves the parking orbit to reach the sphere of influence, and have the same velocity vector that results from the solution of Lambert's problem. Using this time (T_0), the launch azimuth (ψ) and the state vector at burnout time (T_ℓ) are found and printed. Next a two-body solution is used to get from T_ℓ to T_1, and then the payload weight at T_1, yaw, and pitch are calculated and printed. Now the main program is entered using the state vector at T_1 as initial conditions.

C. LAMBERT'S PROBLEM SOLUTION

Given

R_e, $\dot{R}_e$ State of Earth at Julian Date

R_T, $\dot{R}_T$ State of target at Julian Date plus flight time

To find $\dot{R}_e^+$, $\dot{R}_T^-$ and δv_1

$\dot{R}_e^+$ is velocity vector of vehicle at Earth

$\dot{R}_T^-$ is velocity vector of vehicle at target

δv_1 is delta v at the Earth

ΔT is the flight time

$$\delta = \cos^{-1} \frac{R_e \cdot R_T}{r_e r_T}, \qquad r_e = |R_e|, \quad r_T = |R_T|$$

$$\eta = \sqrt{2 r_e r_T} \cos \frac{\delta}{2}$$

There are two solutions:

$$0 \leq \delta \leq 180°$$

$$180° \leq \delta \leq 360°$$

For both solutions, do the following:

$$\omega = r_e + r_T - \sqrt{2}\, \eta F_0 \left(\frac{\alpha}{4}\right)$$

Solve by Newton's method

$$\alpha_i = \alpha_{i-1} + \frac{\sqrt{\mu_s}\,\wedge T - T}{D}$$

$$\alpha_o = 0$$

where

$$T = \omega^{3/2}\frac{F_3(\alpha)}{F_2^{3/2}(\alpha)} + \eta\omega^{1/2}$$

$$D = \frac{3}{8}\omega^{1/2}\eta\frac{F_3(\alpha)}{F_2(\alpha)} + \frac{1}{8}\eta^2\frac{F_2^{1/2}(\alpha)}{\omega^{1/2}}$$

$$+ \frac{\omega^{3/2}}{F_2^{5/2}(\alpha)}\left\{\frac{3}{4}\left[F_5(\alpha) + F_3^2(\alpha)\right] - \frac{1}{4}F_4(\alpha) - \frac{1}{2}F_4(\alpha)F_2(\alpha)\right\}$$

$$F_5(\alpha) = \sum_{i=0}^{\infty}\frac{(-\alpha)^i}{(2i+3)!}$$

$$F_4(\alpha) = \sum_{i=0}^{\infty}\frac{(-\alpha)^i}{(2i+2)!}$$

$$F_3(\alpha) = \frac{1}{6} - \alpha F_5(\alpha)$$

$$F_2(\alpha) = \frac{1}{2} - \alpha F_4(\alpha)$$

$$F_1(\alpha) = 1 - \alpha F_3(\alpha)$$

$$F_0(\alpha) = 1 - \alpha F_2(\alpha)$$

If $\omega < 0$ on any iteration, let

$$\alpha_i = -4 \left\{ \ell n \left[L + (L^2 - 1)^{1/2} \right] \right\}^{1/2}$$

where

$$L = \frac{r_e + r_T}{\sqrt{2\eta}}$$

(should occur for $\eta > 0$) and continue iterating.

If $\alpha_i > 4\pi^2$ on any iteration, let

$$\alpha_i = \frac{\alpha_{i-1} + 4\pi^2}{2}$$

and continue. Iteration is complete when either

$$|\Delta\alpha_i| < |\alpha_{i-1}| \; 10^{-15},$$

where

$$\alpha_{i-1} = 0, \; |\Delta\alpha_i| \leq 1. \times 10^{-15}$$

or

$$|\Delta\alpha_i| < \frac{\sqrt{\mu_s}\ \Delta T}{|D|} \times 10^{-15}$$

Now

$$\dot{R}_e^+ = \frac{1}{g}(R_T - f\,R_e)$$

$$\dot{R}_T^- = \frac{1}{g}(R_e - \dot{g}\,R_T)$$

where

$$f = 1 - \frac{\omega}{r_e}, \qquad g = \frac{\eta\omega^{1/2}}{\sqrt{\mu_s}}$$

$$\dot{g} = 1 - \frac{\omega}{r_T}$$

Now

$$\delta v_i = |\dot{R}_e^+ - \dot{R}_e|$$

$$\delta v_2 = |\dot{R}_T - \dot{R}_T^-|$$

We choose the solution having δ such that $(\delta v_1 + \delta v_2)$ is a minimum.

End of Lambert's Problem

To get to the Earth's sphere of influence, $\Delta t = r_{(ter)}/\delta v_1$ is the time it takes to reach the Earth's sphere of influence and $r_{(ter)}$ is the radius of the Earth's sphere of influence. Now do a two-body solution to get from R_e and $\dot{R}_e^+$ to $R_{(ter)}$ and $\dot{R}_{(ter)}$.

For option (b)

R_o is position of the launch site and a function of t_0 (see Section H of ITEM manual).

$R_p(t_1)$ is the position on the parking orbit at burn time.

r_p is $|R_p(t_1)|$

$\dot{R}_{(ter)}$ is velocity vector at the sphere of influence of the Earth on the conic, from Lambert's problem.

v is $|\dot{R}_{(ter)}|$, $r_{(ter)}$ is input.

t_1 is time of leaving the parking orbit.

$\Delta t\ell$ is time of burn from launch pad to parking orbit.

$\delta\ell$ is central ascent angle.

ψ is launch azimuth.

$\hat{E}$ is unit east vector.

$\hat{N}$ is unit north vector.

ω_E is hourly sidereal rate.

W_1^- is payload after launch - lb.

W_p is final weight - lb.

W_{KD} is power plant weight - lb.

W_{KF} is fuel weight - lb.

W_2^- is $W_p + W_{KD} + W_{KF}$

W_2^+ is $W_p + W_{KD}$

W1D is second stage power plant - lb.

W_1^+ is $W_2^- + W1D$

$\dot{W}_1$ is weight flow for c_1.

W_2 is weight flow for c_2.

c_1 is c for first burn out of parking orbit.

c_2 is c for second burn out of parking orbit.

$t_{(ter)}$ is time at Earth's sphere of influence after Lambert's problem solution.

The iteration equation for t_0 is:

$$t_{0_i} = t_{0_{i-1}} - \frac{\left(t_{0_{i-1}} - t_1 + \Delta t\ell + \dfrac{\delta_{i-1}}{\sqrt{\mu_e / r_p^3}}\right)}{\dfrac{\omega_E}{\sqrt{\mu_e / r_p^3}} \dfrac{(n'd - d'n)}{(n^2 + d^2)} + 1}$$

where

$$\frac{1}{a} = \frac{2}{r_{(ter)}} - \frac{v^2}{\mu_e}$$

$$\alpha_1 = (1 - r_{(ter)}/a) \Big/ (1 - r_p/a)$$

$$\beta = \sqrt{|a|}\,\ell n \left(\alpha_1 + \sqrt{\alpha_1^2 - 1}\right)$$

$$\alpha = \beta^2/a$$

$$G_1 = \beta F_1(\alpha)$$

$$G_2 = \beta^2 F_2(\alpha)$$

$$t_1 = t_{(ter)} - \frac{1}{\sqrt{\mu_e}}(G_3 + r_p G_1)$$

$$h = \sqrt{\mu_e r_p (2 - r_p/a)}$$

$$a_2 = -\frac{h}{v^2}\left(1 - \frac{G_2}{r_{(ter)}}\right)$$

$$a_1 = \left[-\frac{r_p}{\sqrt{\mu_e}} - \frac{a_2\sqrt{\mu_e}}{h}\left(1 - \frac{r_p}{a}\right)\right] G_1$$

$$H = \left[\hat{R}_o \times \dot{R}_{(ter)}\right] * \text{sign } H_z$$

$$n = a_1 \, |H| + a_2 \hat{R}_o \cdot \dot{R}_{(ter)}$$

$$d = a_1 \hat{R}_o \cdot \dot{R}_{(ter)} - a_2 \, |H|$$

$$n' = a_1 \left[\hat{E} \times \dot{R}_{(ter)}\right] \cdot \hat{H} + a_2 \hat{E} \cdot \dot{R}_{(ter)}$$

$$d' = a_1 \hat{E} \cdot \dot{R}_{(ter)} - a_2 \left[\hat{E} \times \dot{R}_{(ter)}\right] \cdot \hat{H}$$

$$\hat{E} = \frac{\hat{k} \times \hat{R}_o}{|\hat{k} \times \hat{R}_o|}, \qquad \hat{k} = \begin{matrix} 0 \\ 0 \\ 1 \end{matrix}$$

$$\delta_i = -\delta\ell + \tan^{-1}\frac{n}{d}$$

Now

$$R_p(t_1) = a_1\dot{R}_{(ter)} + a_2\left[\hat{H} \times \dot{R}_{(ter)}\right]$$

$$\dot{R}_p^-(t_1) = \sqrt{\frac{\mu_e}{r_p^3}}\left[\hat{H} \times R_p\right]$$

$$\dot{R}_p^+(t_1) = \sqrt{2 - \frac{r_p}{a}} \cdot \dot{R}_p(t_1)$$

The quantities are used as initial conditions for a trajectory starting in Earth reference on the parking orbit. Now

$$\hat{N} = \hat{R}_o \times \hat{E}$$

$$v_1 = \sqrt{\mu_e / r_p}$$

$$\psi = \tan^{-1}\frac{\hat{E} \cdot \hat{H} \times R_o}{\hat{N} \cdot \hat{H} \times R_o}$$

$$\hat{R}_o = \hat{N}\cos\psi + \hat{E}\sin\psi$$

$$R_{t\ell} = (\hat{R}_o \cos\delta\ell + \hat{\dot{R}}\sin\delta\ell)\, r_p$$

$$\dot{R}_{t\ell} = (\hat{\dot{R}}_o \cos\delta\ell - \hat{R}_o \sin\delta\ell)\, v_1$$

The last two equations give the positions and velocities on the parking orbit at burnout.

Now to find the payload needed after launch, pitch and yaw:

$$\delta v_1 = \left| \dot{R}_p^+ - \dot{R}_p^- \right|$$

$$W_1^- = W_1^+ \ell \frac{\delta v_1}{c_1} \left(\frac{W_2^+}{W_2^-} \right)^{c_1 \cdot c_2} \quad \text{(payload necessary)}$$

$$\Delta t_1 = \frac{W_1^+ - W_1^-}{W_1}$$

$$\Delta t_2 = \frac{W_2^+ - W_2^-}{\dot{W}_2}$$

$$\Delta t = \Delta t_1 + \Delta t_2$$

$$t_2 = t_1 + \Delta t$$

Do a two-body solution from t_1 to t_2 on the conic which takes the vehicle to the Earth's sphere of influence.

$$S_o = \frac{\left[R_p(t_1) + R(t_2) \right]}{2}$$

$$s = |S|$$

$$T = \dot{R}(t_2) - \dot{R}(t_1) + \mu_e \frac{\Delta t}{s^3} S_o$$

$$\text{Yaw} = \sin^{-1} (\hat{H} \cdot \hat{T})$$

$$\text{Pitch} = \tan^{-1} \left[\frac{\hat{H} \times \hat{R}_p(t_1) \cdot \hat{T}}{\hat{R}_p(t_1) \cdot \hat{T}} \right]$$

Table II-1

Sample Input For Run Starting In Parking Orbit

DATA PROCESSING LAYOUT SHEET - 80 COLUMNS

```
         4  ID FOR LAUNCH DATE AND TARGET NO.
 1974    5   21    0    0  0,0    3    6
         9  ID FOR MAXIMUM TIME AND NO. OF PLANETS
    0.0      0.0     17680     0.0       0.0       0.0       6.
        14  ID TO TRIGGER PERIGEE PRINT
         0         0         1
        11  SETS INTEGRATION AND PRINT INTERVALS
   1.     4.       150.
 .03125   .5
   1.     6.
   .27    10.
   .01
   1.
   .1     10.
   .005
   480.
    0.0   10.
 .03125
   24.
    0.0   10.
 .03125
   24.
   0.0    .5
  .0005
   240.
```

Table II-1 (Continued)

DATA PROCESSING LAYOUT SHEET - 80 COLUMNS

```
        16  RADAR INFORMATION
         2
 ROSMAN
  277.    7.         4.2       35.      12.        .7        2874.56
 CANBERRA
  148.    57.       20.9      -35.      37.       52.7       3766.588
         3  CONTROL OPTIONS
2111
         5  LAUNCH SITE INPUT
  -60.     26.5       .5       7.7       7.91         0.      1000.
         7  LAMBERT OPTION
 2188.    550.       3.        123.3
    0  0.0            .51373647D-06
    0.0
    0.0
 -.25     1000000.    0.0       0.0
   20.     444.     290.      21650.     3000.    48.76     10.3448
   500.    476.     1440.     3750.      .33333333 12.       1.025
         3  CONTROL OPTIONS FOR SPECIAL PRINTS AND BETA MODE
2111  1  1                      1                                      1
        20  ID TO START INTEGRATION
/*
```

APPENDIX III

STAFF PAPER NO. 69-2

LAUNCH VEHICLE CONSIDERATIONS FOR DEVELOPING AN OUTER PLANETS EXPLORATION STRATEGY

by

George M. Levin
Advanced Plans Staff
Goddard Space Flight Center

February 1969

NOTE: The data presented in this Staff Paper are derived from sources that are considered to be sufficiently accurate for advanced planning. In no instance should these data be used for detailed mission planning.

INTRODUCTION

In developing a strategy for the exploration of the outer planets (Jupiter, Saturn, Uranus, Neptune, and Pluto) one must take into consideration a multitude of factors. These factors include mission cost, flight time, spacecraft complexity, science complement required, and related considerations. One of the major considerations is the trade-offs which can be made in the areas of launch vehicle cost, launch vehicle payload, and flight time. The purpose of this paper is to indicate these trade-offs.

To begin this analysis, one must first assess the payload capability of the various launch vehicles that will be available in the 1970's. These launch vehicles exhibit certain characteristics which allow them to be categorized as small, medium or large. These characteristics are cost and payload. Since all things are relative, the following definitions shall be applied to the terms small, medium, and large.

	Small	Medium	Large
Launch Vehicle Cost	Less than 13 M	13 M to 20 M	Greater than 20 M
Class of Payload	Less than 700 #	700 - 1200 #	Greater than 1200 #

LAUNCH VEHICLE PERFORMANCE

Figures III-1 and III-2 show the payload weight versus characteristic velocity for those launch vehicles which are currently being considered for the 1970's. In Figure III-1, the kick stage (or velocity package) shown is the Burner II (2336). This is a growth version of the present attitude-stabilized Burner II configuration. It assumes a larger (2336 pounds) loading of the current Burner II propellant.

In Figure III-2, the kick stage shown is the TE-364-4. This is a growth version of the spin-stabilized TE-364 Thiokol series solid-propellant motors. Its characteristics are:

<u>TE 364-4</u>

Gross motor weight	2,244 lb.
Weight at "all burn"	129 lb.
Total Impulse	602,400 lb/sec.
Specific Impulse	287 sec.

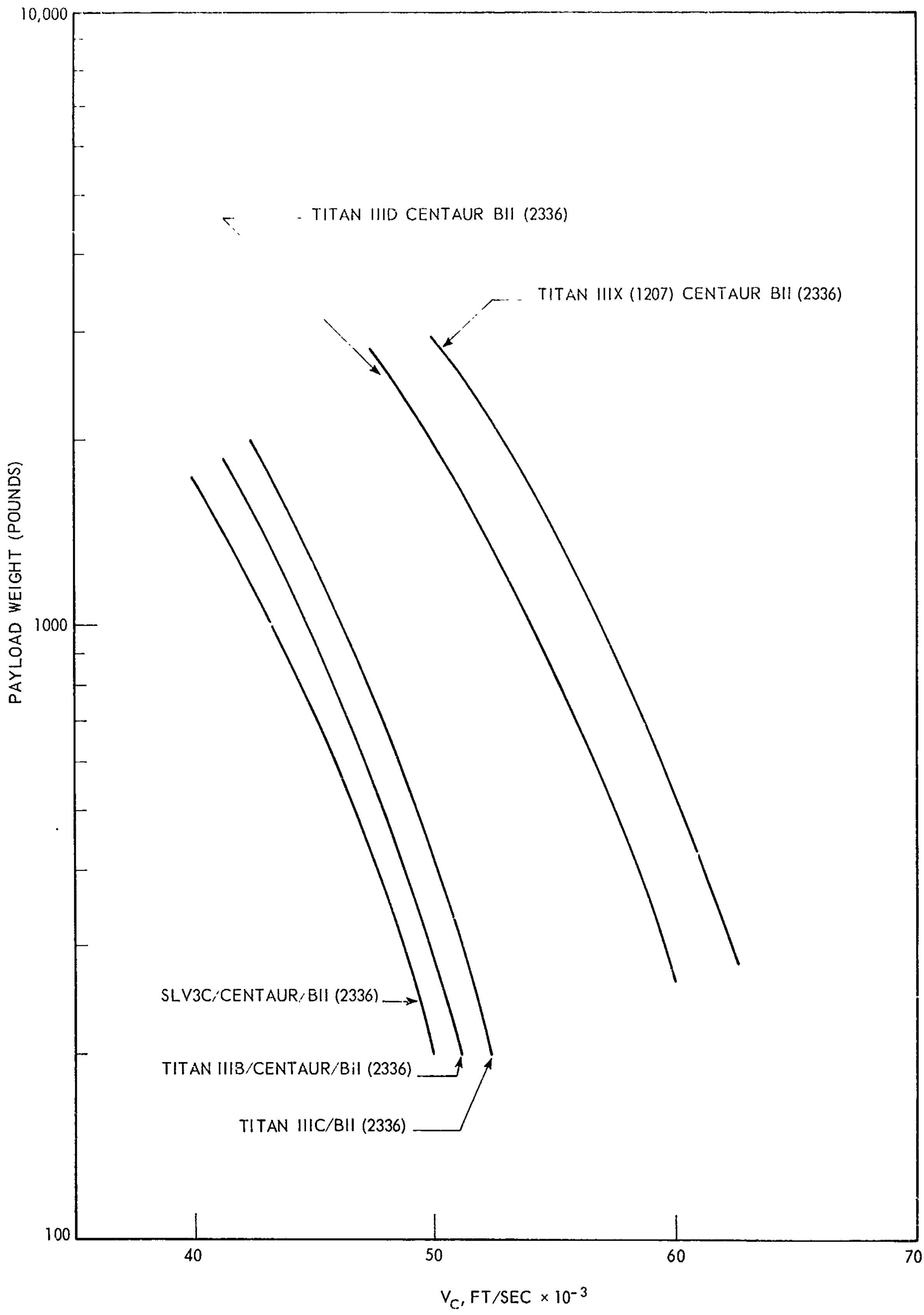

Figure III-1. 1969-73 Possible Launch Vehicle Performance with Burner II (2336) Velocity Package (Ref. 1)

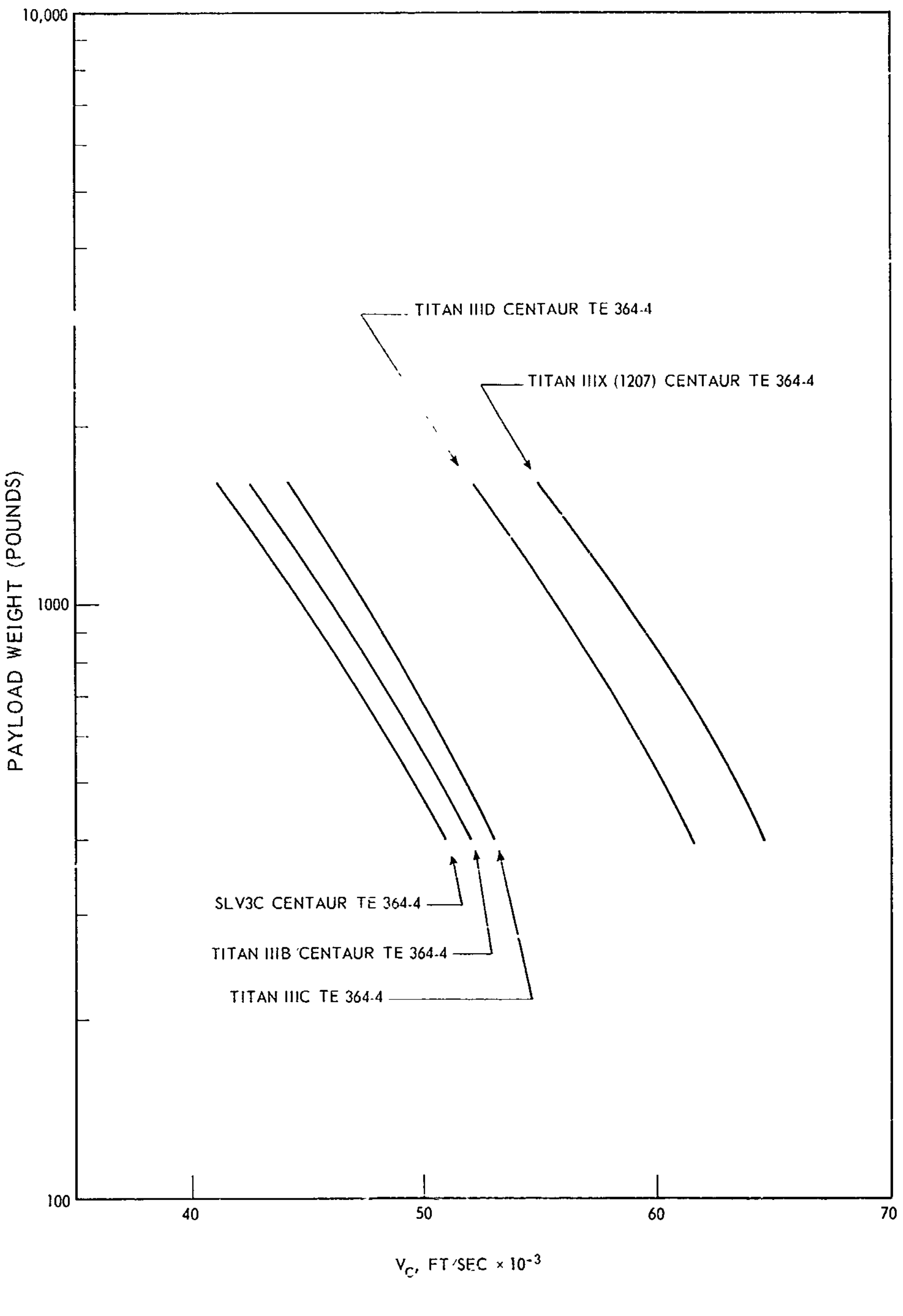

Figure III-2. 1969-73 Possible Launch Vehicle Performance with TE 364-4 Velocity Package (Ref. 1)

Recent changes in the nomenclature of the launch vehicles noted in Figures III-1 and III-2 as well as a description of these vehicles is included in Table III-1.

Table 1

Launch Vehicle Nomenclature

Name	Description
SLV3C	Atlas launch vehicle designed for use with Centaur upper stage.
TITAN IIIB or TITAN IIIX	Basic two-stage Titan core.
TITAN IIIC	Basic core with two 5- segment, 120-inch solid propellant motors as the zero stage and the Titan transtage as the third stage.
TITAN IIID or TITAN IIIX (1205)	Basic two-stage Titan core with two 5-segment, 120-inch solid propellant motors as the zero stage.
TITAN IIIX (1207)	Improved Titan core with two 7-segment, 120-inch solid propellant motors as the zero stage.

Note: Care must be exercised when using the data presented in Figures III-1 and III-2. These data are sufficiently accurate for advanced planning purposes. When comparing data presented in Reference 1 and similar data from other sources, it is not uncommon to detect variations in payload of 10 percent for a given vehicle at a given characteristic velocity.

COST

One of the major factors that influences the overall cost of a project is the choice of a launch vehicle. Whether a specific launch vehicle is developed is dependent upon establishment of a need for the vehicle. And finally, the cost of the launch vehicle is influenced by its use rate. The SLV3C/CENTAUR and TITAN IIIC have already been developed and are in use. The TITAN IIID/CENTAUR is being developed for the Mariner Mars 1973 Mission. The TITAN IIIB/CENTAUR is the same basic vehicle as the TITAN IIID/CENTAUR without the two 5-segment, 120-inch solid propellant motors. In addition, dropping the solid propellant motors may necessitate minor changes in the guidance package. However, the important point to note here is that if the TITAN IIID/CENTAUR is developed then one can assume that the TITAN IIIB/CENTAUR will be developed. The TITAN IIIX(1207)/CENTAUR development is contingent upon the planned development of the TITAN IIIX(1207) vehicle by the Air Force.

Table III-2 ~~lists~~ the costs of the various launch vehicles that have been discussed. These cost data were obtained from the Advanced Programs and Technology Division of the Launch Vehicle and Propulsion Programs Office at NASA Headquarters. Since the final launch vehicle costs are strongly dependent on use rate, these cost data should be considered representative and used for comparison only.

Table 2

Cost of Various Launch Vehicles

Name	Cost
SLV3C/CENTAUR	10.0 M
TITAN IIIB/CENTAUR	11.8 M
TITAN IIIC	17.2 M
TITAN IIID/CENTAUR	17.4 M
TITAN IIIX (1207)/CENTAUR	24.4 M

The development cost of the TE-364-4 from the existing TE-364-3 has been estimated at approximately $1.5 million. Data from the Advanced Programs and Technology Division estimates the procurement cost of the TE-364-4 at 100 K each. Thus, the additional cost of the TE-364-4 velocity package is insignificant when compared to the cost of any of the launch vehicles.

LAUNCH ENERGY AND FLIGHT TIME REQUIRED FOR OUTER PLANETS EXPLORATION

Figures III-3 and III-4 show the characteristic velocity required for probing the outer planets with and without Jupiter swingby respectively. Figures III-5, III-6, and III-7 are more detailed comparisons of two-planet swingby and direct flight to Saturn, Uranus, and Neptune respectively. Figures III-3 and III-4 are from Reference 1, while Figures III-5, III-6, and III-7 are from Reference 2.

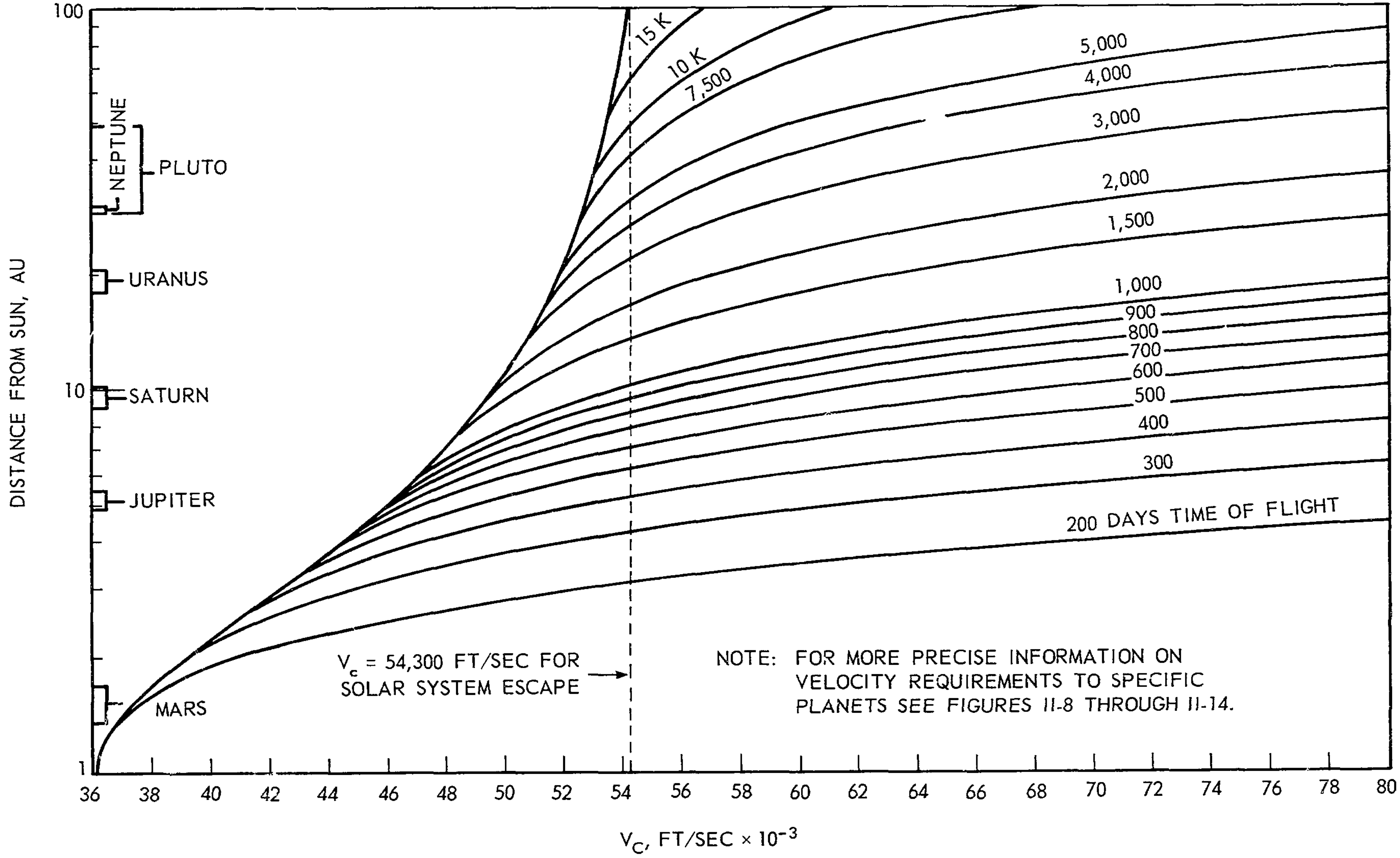

Figure III-3. Velocity Required for Ballistic Probes to Outer Planetary Regions (Ref. 1)

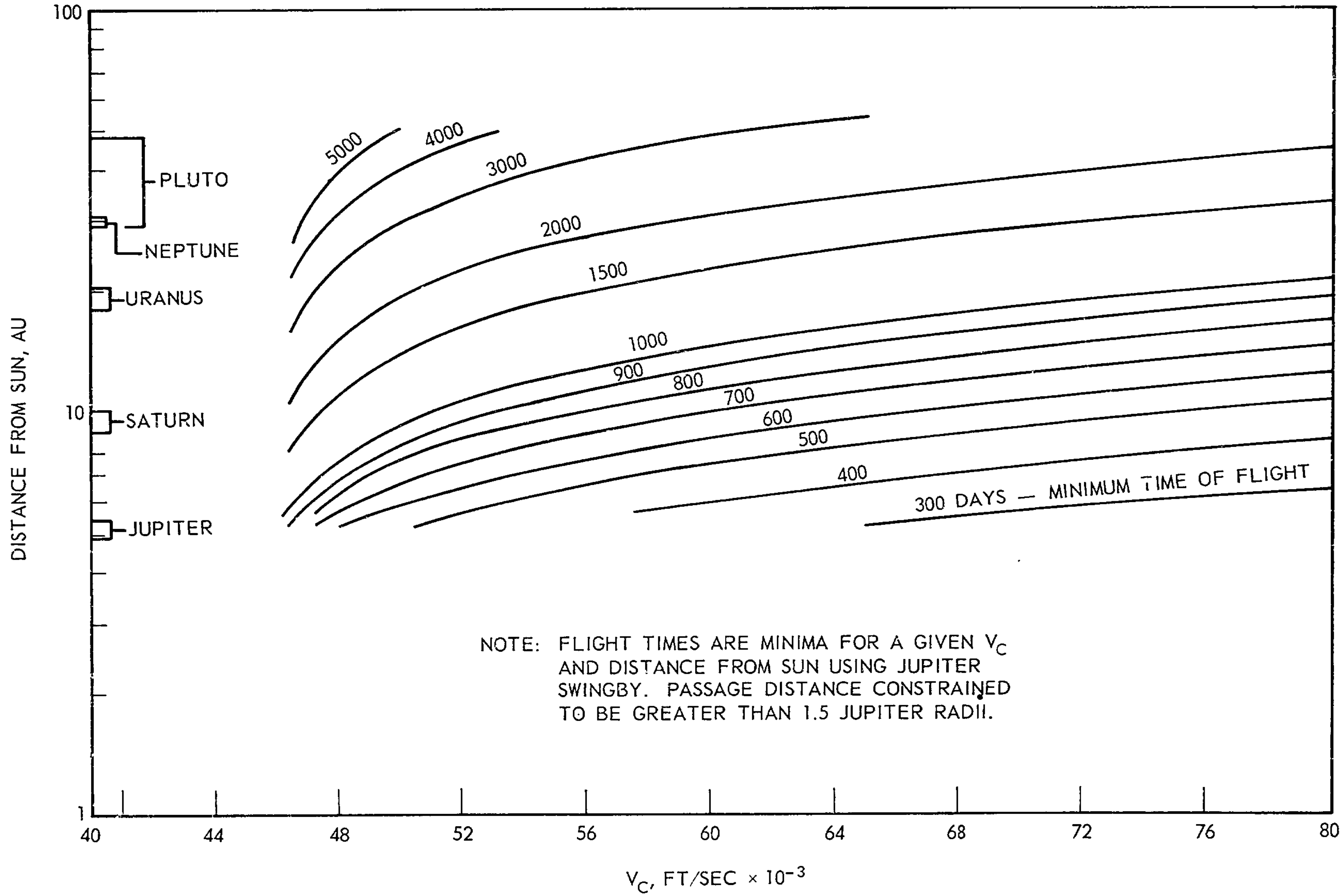

Figure III-4. Velocity Required for Probes to Outer Planetary Regions with Jupiter Swingby (Ref. 1)

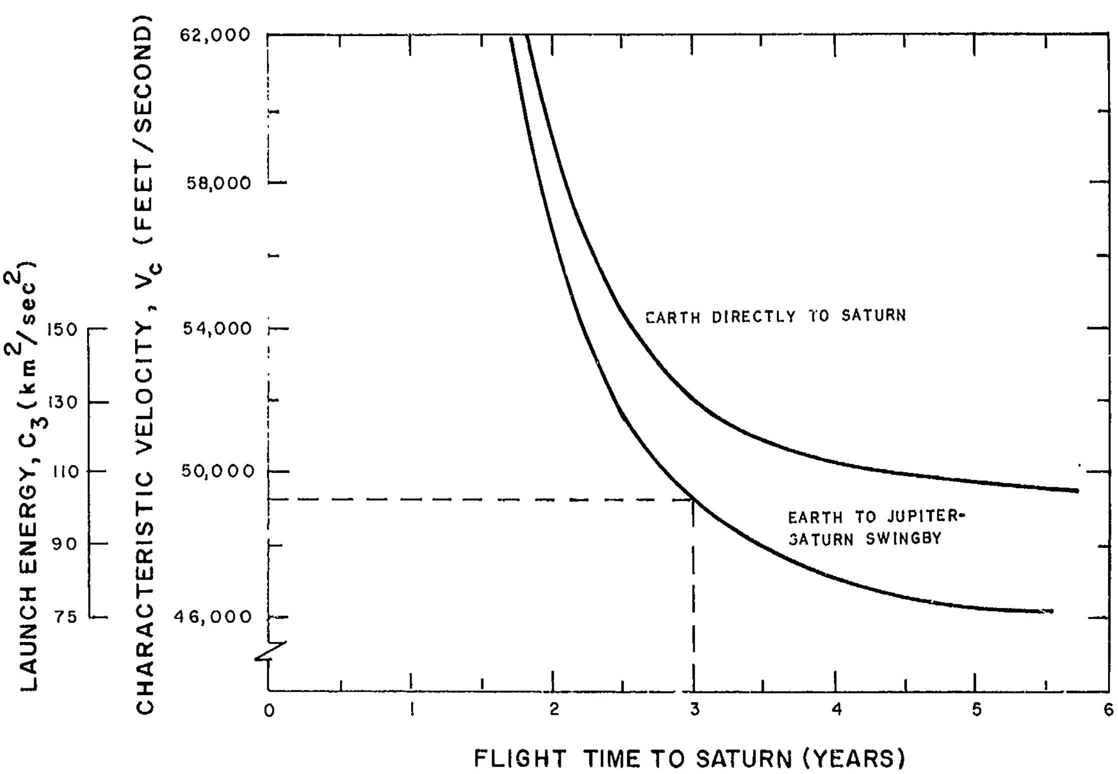

Figure III-5. Two-Planet Swingby and Direct Flight to Saturn (Ref. 2)

One of the points to be noted from these figures is that using a Jupiter flyby it is possible to reach Saturn in 3 years (1,100 days) with a characteristic velocity of 49,200 ft/sec. The flight time to Jupiter is in the order of 550 days at this velocity.

In the case of the Saturn, Uranus, and Neptune missions, the opportunity for using a Jupiter swingby occur only at the frequency of the synodic period between Jupiter and the target planet, although these opportunities last for 3 to 5 years. The opportunity then ends when Jupiter moves ahead of the outer planets. Subsequent opportunities for these missions occur in regular cycles approximately as shown:

Jupiter-Saturn	1976-1980 and then 1996-2000 and so on
Jupiter-Uranus	1978-1982 and then 1992-1996 and so on
Jupiter-Neptune	1978-1982 and then 1991-1995 and so on

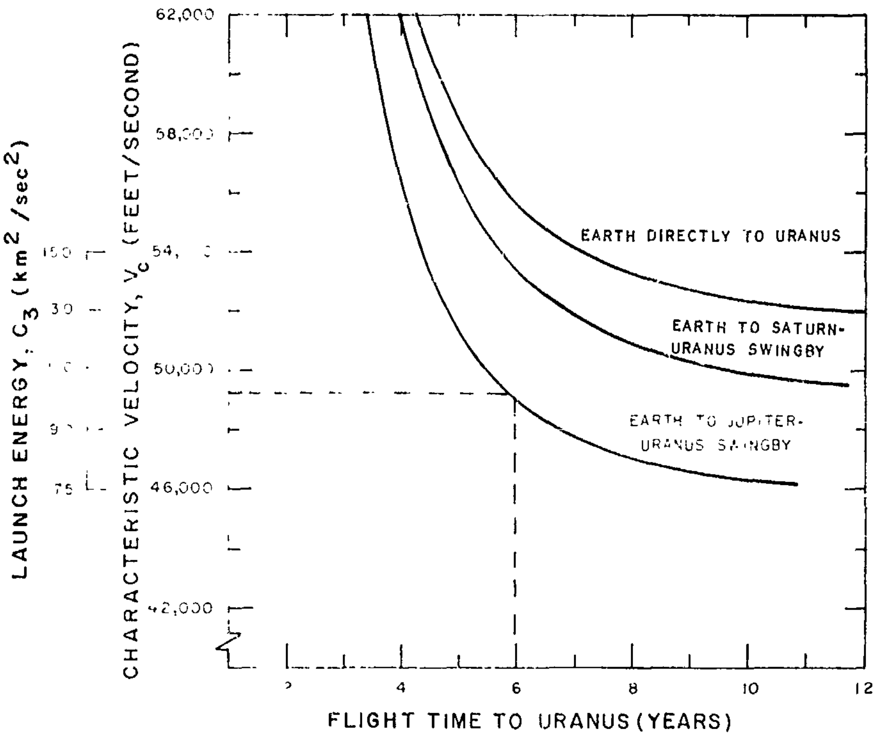

Figure III-6. Two-Planet Swingby and Direct Flight to Uranus (Ref. 2)

SPACECRAFT

Reference 3 describes a 600 pound class spin-stabilized spacecraft with an experiment weight of 50 pounds. This basic spacecraft with minor modifications is capable of performing three types of missions (Reference 2). These are:

1. Jupiter Flyby Mission

2. Out-of-Ecliptic Mission

3. Two Planet (Jupiter-Saturn) Swingby.

Reference 4 describes a 1,000 pound class spin-stabilized spacecraft with an experiment weight of 200 pounds. This spacecraft is capable of performing the same missions as the 600 pound Galactic-Jupiter Probe as well as additional outer planetary exploration missions. The so-called "Outer Planets Explorer (OPE)" can perform the following missions:

1. Jupiter Flyby Mission

2. Out-of-Ecliptic Mission

3. Two Planet Swingbys

   a. Jupiter-Saturn
   b. Jupiter-Uranus
   c. Jupiter-Neptune
   d. Jupiter-Pluto
   e. Saturn-Uranus
   f. Saturn-Neptune
   g. Saturn-Pluto

4. Three Planet Swingby

   a. Jupiter-Uranus-Neptune

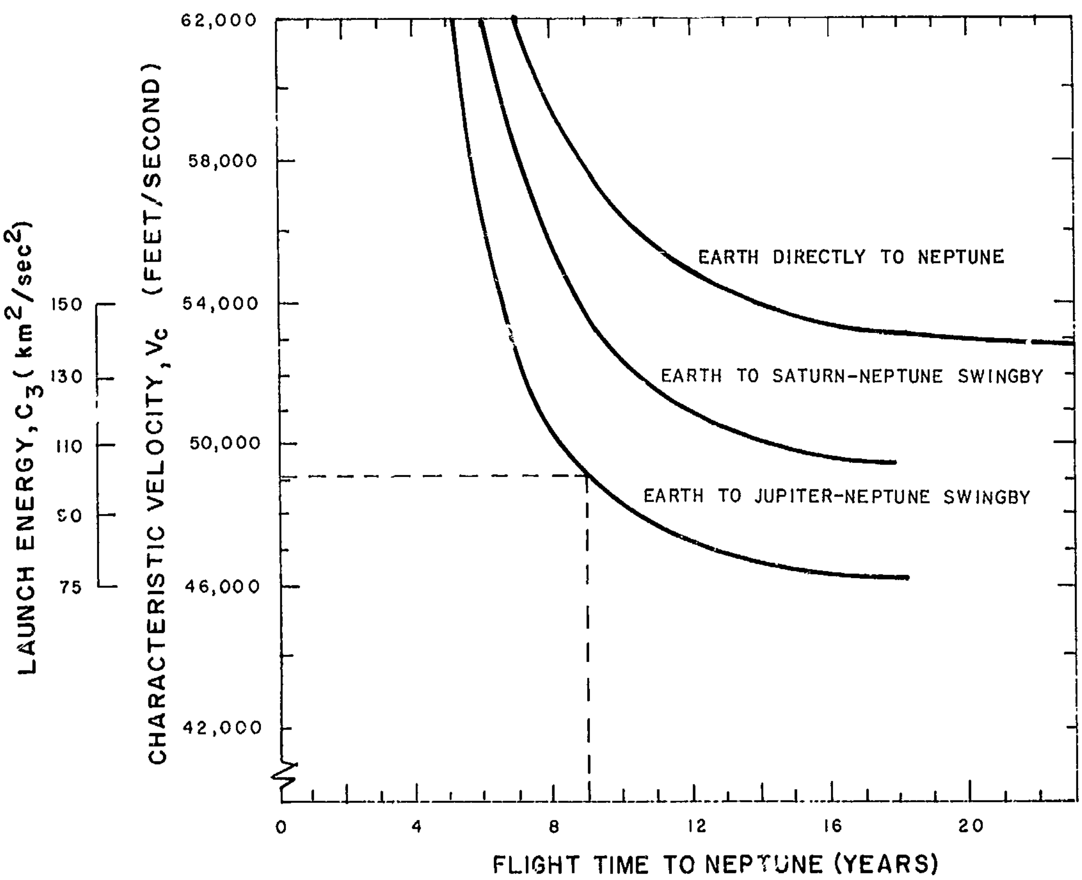

Figure III-7. Two-Planet Swingby and Direct Flight to Neptune (Ref. 2)

CONCLUSION

Data from Reference 2 indicates that the smaller type (600 pound Galactic-Jupiter Probe) is capable of performing the three so-called first-step missions, namely:

a. the Jupiter Flyby Mission

b. the Out-of-Ecliptic Mission

c. the Jupiter-Saturn Swingby

Furthermore, the data from Reference 1 (Figure 2) shows that the SLV3C/CENTAUR/TE-364-4 or the TITAN IIIB/CENTAUR/TE-364-4 is capable of providing the characteristic velocity required to perform one or more of the above three missions. The important factor here is that it is possible to combine the smaller, lower-cost spacecraft with the smaller, lower-cost launch vehicles and still perform a significant outer planetary exploration.

While the early studies indicate that two planet missions may be accomplished with a six hundred pound spacecraft, it is recognized that such a mission would be primarily restricted to interplanetary and precursory planetary science. A larger class spacecraft must be employed to achieve further desired detailed planetary scientific objectives (Reference 5).

REFERENCES

1. "NASA Launch Vehicle Estimating Factors, January 1969."

2. Task I Report, Analysis of Selected Deep-Space Missions for NASA Evaluation with Models of Optimized Nuclear Spacecraft, Goddard Space Flight Center No. X-701-69-170, in this Report.

3. "Galactic Jupiter Probe, Phase A Report," GSFC X701-67-566, November 1967.

4. "GSFC Presentation to NASA Hq. Symposium on Outer Planets Exploration", January 1969.

5. GSFC Document X-701-69-189 covering Outer Planets Explorer.

APPENDIX IV

A STRATEGY FOR EXPLORATION OF THE OUTER PLANETS

"But it is clear that, for the second decade in space, our previous experience and success permit us to concentrate more heavily on the goals of space exploration rather than its uncertainties and difficulties. Furthermore, as we develop programs of space exploration, their interest and value to the nation can be greatly enhanced by planning the programs to maximize their scientific return." *

One strategy for obtaining scientific information covered in this report would employ the Grand Tour mission. The Grand Tour mission when described as a sequential flyby of four planets in very restricted in terms of launch opportunities and is very demanding on spacecraft subsystems requirements, particularly guidance. Another strategy for obtaining the desired scientific information would employ two planet swingby missions as described in this Report. These missions provide frequent launch opportunities while not too demanding on spacecraft systems. An elaboration of this strategy has been discussed among advanced mission planners and management personnel at Goddard Space Flight Center and is shown in simplified form in Table IV-1 without including funding requirements. W. G. Stroud has summarized this strategy by stating:

"This is a program, in contrast to a project, consisting of 6 to 8 flights of a basic spacecraft carrying scientific instruments for both interplanetary measurements and the outer planets and their environments. This spacecraft is to be small as possible and as inexpensive as possible, consistent with the scientific and technological requirements. A spin-stabilized spacecraft, a growth version of the Galactic Probe and the Galactic Jupiter Probe previously proposed, with a despun platform for planetary imagery is planned. (This growth version can be considered as the OPE. Ed.)

"A basic element of this approach is that the very long flight times, upwards of eight years, require a high order of redundancy in the scientific payload, and therefore a sizeable fraction of the spacecraft weight assigned to that payload." †

* "The Space Program in the Post Apollo Period", a Report of the President's Science Advisory Committee, Feb. 1967, Pg. 8.

† Internal memorandum by W. C. Stroud, dated Feb. 24, 1969, Subject: Guidance on the Strategy for Exploration of Outer Planets.

Table IV-1

Simplified Presentation of a Strategy for Exploration of the Outer Planets

Year Launched	Mission	No. of Prototype Spacecraft	No. and Weight of Flight Spacecraft (lbs.)	Weight of Scientific Instruments (lbs.)	Estimated Operational Lifetime of Spacecraft (yrs.)
74	J	1	1 (750)	100	2 1/2 - 3
75	J O/E	-	1 (750)	100	2 1/2 - 5
76	-	-	-	-	-
77	J-S	1/2	1 (1000)	200	2 1/2 - 3
77	J-P	1/2	1 (1000)	200	7 - 8
78	J-S	-	1 (1000)	200	2 1/2 - 3
78	J-P	-	1 (1000)	200	7 - 8
79	J-U	1/2	1 (1000)	200	4 - 5
80	J-N	-	1 (1000)	200	7 - 8
TOTALS		2 1/2	8		

Mission Code:

J	=	to Jupiter and beyond
J O/E	=	to Jupiter thence out of ecliptic plane
J - S	=	to Jupiter, thence to Saturn (and beyond or impact)
J - P	=	to Jupiter, thence to Saturn (and beyond or impact)
J - U	=	to Jupiter, thence to Uranus (and beyond or impact)
J - N	=	to Jupiter, thence to Neptune (and beyond or impact)